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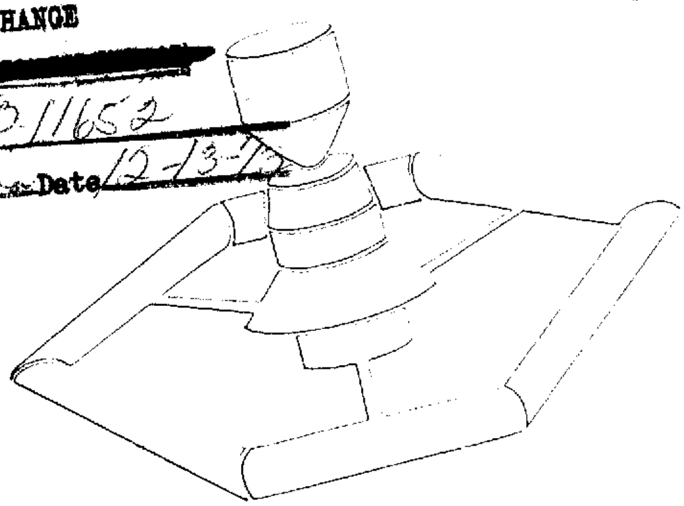
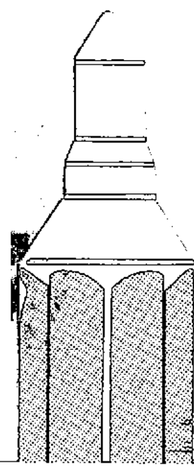


SELF-DEPLOYING SPACE STATION

INTERIM SUMMARY REPORT



(NASA-CR-55597) SELF-DEPLOYING SPACE STATION, VOLUME 1 Interim Summary Report (North American Aviation, Inc.) N74-71035
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NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION

FOREWORD

This document was prepared by the Space and Information Systems Division of North American Aviation, Inc., under the National Aeronautics and Space Administration Contract NAS1-1630, "Self-Erecting Manned Space Laboratory," dated 27 October 1961.

Because of the information relative to physical characteristics of the Apollo vehicle contained herein, this document has been classifie[redacted]. All other information presented is unclassified.

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I. INTRODUCTION

The Space and Information Systems Division of North American Aviation, Inc., was awarded a 6-month study contract, NAS1-1630, on 27 October 1961 by the National Aeronautics and Space Administration, Langley Research Center (LRC), Langley Air Force Base, Virginia, to examine the feasibility of a self-deploying space station. Through this study, S&ID has established the major characteristics of an "optimized" space station design embodying the self-deploying principle. The basic concept, developed initially by LRC, has been adhered to throughout the study, although changes were incorporated when advantages of such modifications were shown.

During the study, a number of space station configurations which employ the basic NASA design concept were investigated. Several critical technical problem areas have been identified and examined in detail to determine the best possible solution. The requirements for the various space station subsystems have been defined, representative subsystems have been examined for possible application, and a preferred concept has been selected and analyzed in considerable detail. Brief studies of development factors, operational costs, and logistics associated with the space station operation were conducted in the latter portion of the study period.

S&ID was recently awarded a 5-month contract extension to design and construct a one-tenth scale model of the "selected" space station configuration that could be used to study the kinematics of the vehicle deployment. As a result, this report has been designated an interim report; the final report, covering all contracted activities, will be submitted to NASA in September 1962. An earlier midterm report, SID 62-191, which covers the results of the first 3 month's studies, was submitted to NASA on 2 February 1962.

The Astro Research Corporation (of Santa Barbara, California), as a funded subcontractor to S&ID on this study, was responsible for the analysis and design of the inflatable material which was used in the initial space station designs. In other technical areas, the S&ID analyses were conducted in sufficient detail to firmly establish system concepts and to determine feasibility; but to provide a more detailed information in certain of these areas, a number of other companies provided data on suggested system designs (based on the S&ID established requirements). The participating companies, and their area of interest, are as follows:

Nutrition and Hygiene

Whirpool Corporation
St. Joseph, Michigan

Environmental Control

Hamilton-Standard, Division of United Aircraft
Windsor Locks, Connecticut

Water Recovery

Ionics, Incorporated
Cambridge, Massachusetts

American Machine and Foundry, Mechanics Research Division
Niles, Illinois

Hamilton-Standard, Division of United Aircraft
Windsor Locks, Connecticut

Communications

Airborne Instruments Laboratory
Dearpark, Long Island, New York

Bendix Systems Division
Ann Arbor, Michigan

Motorola, Incorporated
Military Electronics Division
Scottsdale, Arizona

Sylvania Electronic Systems
Buffalo, New York

Power Supply (Solar Cells)

Hoffman Electronics, Semiconductor Division
El Monte, California

Spectrolab
Los Angeles, California

Radio Corporation of America
Princeton, New Jersey

Texas Instruments, Semiconductor Division
Dallas, Texas

Batteries

Gulton Industries
Metuchen, New Jersey

Sanotone
New York, New York

Yardney
New York, New York

Eagle-Picher
Cincinnati, Ohio

Power Control Equipment

Gulton Industries
Engineering Magnetics Division
Hawthorne, California

Mallory
Long Beach, California

Hamilton-Standard, Division of United Aircraft
Windsor Locks, Connecticut

Electro-Optical Systems
Pasadena, California

International Telegraph and Telephone
New York, New York

This interim report has been divided into the following four major volumes:

1. SID-62-658-1, Summary Report
2. SID-62-658-2, Appendixes

3. SID-62-658-3, Development Planning

4. SID-62-615, Preliminary Stress Analysis

The appendixes volume (SID-62-658-2) has been further subdivided into two parts: (1) An Astro Research Corporation report covering their subcontract activities, and (2) an S&ID report on "Meteoroid Effects on Space Vehicles." Supplementary reports, covering specialized areas studied by the other companies in conjunction with the S&ID study, are being submitted to NASA with the S&ID reports.

II. SUMMARY

The concept of the self-deploying space station is unique in that it incorporates most of characteristics that are highly desirable for an early operational space station. Any space station—whether it be developed for use as a laboratory, as an orbital launch platform, as a communications station, or as a military base of operations—will require a large volume to accommodate equipment and to provide living and working space for crew members. The provisioning of artificial gravity (required at least for experimental purposes) leads to need for large-dimensional configurations that can be slowly rotated.

The self-deploying space station has been designed to be compatible with planned launch vehicles and manned spacecraft. The Saturn C-5 (S-ID and S-II stages only) could be used for boosting the station into a low-altitude (about 300 nautical miles) orbit. One Apollo spacecraft (command module, service module, and escape system) can be launched with the space station, thereby enabling the crew members to monitor the station deployment and to conduct the initial operations aboard the station after deployment. In this role, the Apollo vehicle will function as an emergency escape system and later as a personnel transport and resupply vehicle which can be launched by the Saturn C-1 booster, or equivalent (such as the Titan III). In order to provide the utmost precaution against possible malfunctions of the space station in orbit, a sufficient number of Apollo vehicles will always be docked at the space station to permit immediate escape of all crew members.

On the basis of preliminary data that are available at this time, it has been concluded that a space station diameter equal to or greater than 150 feet is feasible. Human factors considerations indicate that larger diameters generally permit more comfortable living conditions; however, booster payload limitations impose a rigid restriction on maximum diameter. The maximum possible diameter of self-deploying space station that could be launched by the Saturn C-5 booster has not yet been determined.

By varying the design of the station hub, docking provisions can be provided for a relatively large number of Apollo vehicles. Docking facility designs have been established for 2, 7, and 9 Apollo vehicles; however, the final docking facility arrangement will be a function of the number of crew members on the space station and the crew capacity of the Apollo command module. Modifications to the command module may increase the number of crewmen that can be carried on the personnel transport mission. The required space station crew size will probably be of the order of 12 to 15 men, although a fairly comprehensive task analysis must be performed to determine the exact number. About midway through the study, a crew size of 21 was arbitrarily selected in order to establish a design point for providing docking facilities and for sizing various space station systems, such as environmental control and power.

The systems which make up the self-deploying space station are designed for an early operational time period. By utilizing such an approach, NASA will be enabled to launch the space station as soon as a Saturn C-5 becomes available. At the present time, the first launch of the C-5 is expected to be in mid-1965; consequently, all systems are based upon use of components whose reliability will have been well established by that time. This conservative design approach provides increased reliability at the expense of weight. If it should be determined that a lower confidence level is acceptable for the initial operations of the space station, a somewhat lower station weight can be achieved. It has been concluded that using this approach, there are apparently no insoluble problems associated with the development of a self-deploying space station for operation in 1966.

Close liaison was maintained between LRC and S&ID during the study period. Results of parallel studies conducted by the LRC Applied Materials and Physics Division have been integrated continuously with those of S&ID, and the resulting conclusions actually represent the best thinking from both organizations. An excellent example of this working relationship is in the space station configuration itself where the concept of hinged modules was devised by S&ID, but the double hinge which makes the concept practical was designed by NASA.

The recommended space station configuration, illustrated in Figure 1, is composed of six rigid cylindrical modules arranged in a hexagonal shape. The modules are hinged together in a manner that will enable them to be folded into a compact cluster for launch by a single Saturn C-5 booster (S-ID and S-II stages). When deployed, the space station is 150 feet in diameter and is rotated at about 3 rpm to create an artificial gravity of about 0.20 g.

Three rigid telescoping spokes join the modules to a central hub. The spokes maintain the central position of the hub and give the crew members free access between the hub and modules of the space station. In the launch configuration, the spokes are retracted to about half their original length and are stowed in the cavity formed by the module cluster.

The space station is automatically deployed from its launch configuration into its orbital configuration by a series of mechanical screw-jack actuators located on the hinge lines between the modules. The spokes are hinged at each end so that they may follow the motions of the modules and thereby may be extended and locked in place at the completion of deployment.

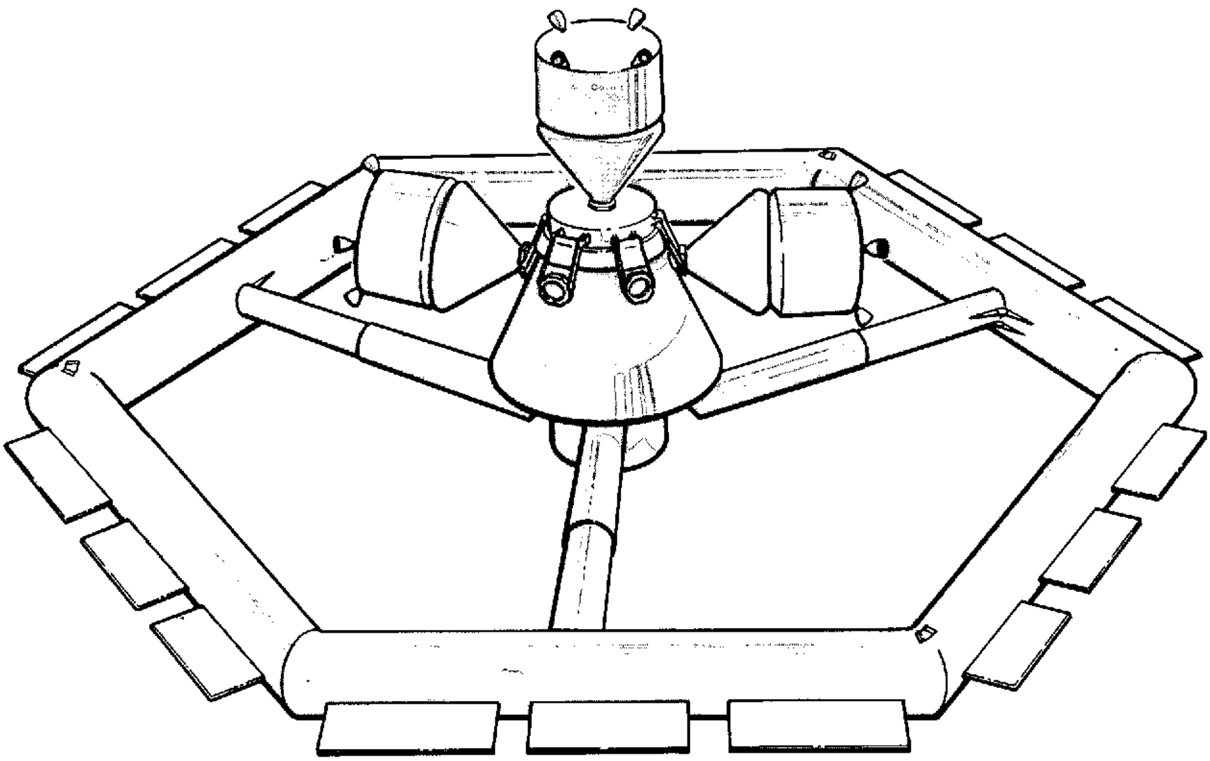


Figure 1. Space Station Configuration

The crew members, who are launched with the space station in an Apollo vehicle, will monitor the deployment but will not enter the station until deployment has been completed.

The six modules are divided between two basic functions—three modules are used for living areas (eating, sleeping, and recreation), and the remaining three are used for mission operations (station operation and control, laboratory experiments, etc.). Each living area module is self-sufficient, containing food-storage, food preparation, and personnel hygiene facilities. Sleeping accommodations are provided for seven crew members. A recreation area for use in off-duty periods is also included. The three work-areas or mission-operation modules contain equipment pertinent to the mission such as experimental research systems, components, and testing equipment. A command center is included in one of the work-area modules for primary space station control and communication. From this center, crew personnel may monitor the activities in all the modules and control the overall operations of the space station.

The structure of the space station has been designed to minimize the number of penetrations that will occur from collisions with meteoroid particles. A multiwall structure composed of three layers of aluminum spaced 1 inch apart and separated by aluminum honeycomb (between the inner layers) and polyurethane foam (between the outer layers) appears to offer adequate protection against nearly all the particles that will be encountered. The walls of the modules, spokes, and hub are made in this same structural configuration.

It is recognized that all equipment, despite its reliability, will at times fail. It is further believed that failures may occur on the space station that would create situations which would force the abandonment of a particular segment, either temporarily or permanently. In order to avoid jeopardizing the mission and the lives of the crew members by such an incident, it is considered necessary to divide the space station into compartments and to install a completely independent environmental and power system in each of the major compartments.

A pressure bulkhead and an airlock have been built between each of the modules and at the ends of each spoke, thereby dividing the space station into 10 compartments. In event of a malfunction or catastrophe in any one of these areas, the affected compartment can be isolated from the other compartments. Crew members can re-enter through the airlocks to conduct repairs.

Duplicate environmental control systems are located in the hub and in each of the modules. The systems in the modules are designed to accommodate six men continuously, or larger numbers for shorter periods of time. Each of these systems performs thermal and atmospheric control functions. Through use of specialized vehicle surface coatings and a radiator, the thermal control system maintains a constant balance between module heating requirements and the heat inputs from the sun, crew members, equipment, etc. The atmospheric control systems work in conjunction with the thermal control system to circulate the air through the cabin; to filter out dust particles; to remove odors, contaminants, and water vapor; to prevent build-up of carbon dioxide; and to maintain the desired partial pressures of oxygen (4 psia) and nitrogen (6 psia). Oxygen regeneration by electrolysis of water is recommended to minimize resupply requirements; however, stored liquid oxygen is available in event of equipment failure. The modules and hub, sealed before launch, contain enough air for the initial pressurization of the entire station. Since the probability of accidental loss of the atmosphere may be fairly high during the early operations, sufficient amounts of oxygen and nitrogen are stored within high-pressure containers to repressurize the complete station two times.

The central hub and each module has a silicon solar cell array externally attached to the structure, thereby providing each of the six modules and the hub with its own power source. Basic power is dc with ac provided by means of inverters and regulated by static regulators. During the portion of the orbit period when the solar cells are exposed to the sun, electrical energy is generated. The size of the solar arrays is sufficient to supply all the electric energy requirements for equipment operation within the space station and, at the same time, to charge nickel-cadmium storage batteries that supply the power for the operation of the space station during the dark period of the orbit.

The normal functions of the space station communication system include station-to-ground, and station-to-resupply vehicle communications; data collection, processing, and storage; and space station intercommunications. It is anticipated that a large amount of data will be transmitted to the earth on a routine basis, particularly if the station is also used to collect meteorological data; consequently, a wide-bandwidth transmitter has been recommended. Equipment has been included for visually monitoring experiments and for controlling docking operations from the space station. Since the space station is continually changing its attitude with respect to the earth, two hemispherical-coverage antennas have been mounted on the space station (one on each end of the hub) to permit communication with earth stations at all times.

A reaction-jet control system, with two nozzles located on the rim of the space station, is used to establish and maintain the desired rotational velocity. The system can be used to increase or decrease rotational rate and it can be controlled at the space station command console. A storable hypergolic bipropellant is used in all the reaction control jets. The silicon solar-cell panels must be oriented to within 10 degrees of the sun to maintain their peak output. A system of four constant thrust-level reaction jets is used to maintain this solar reference. In addition to changing the station attitude by 1 degree per day to account for normal precession, the four jets reorient the station in the event of any external disturbance, such as the impact of Apollo vehicles docking on the hub. Internal disturbances cause only wobble, not permanent disorientation of the spin axis. A precession wheel is normally used to damp (within three revolutions) the space station wobble that results from either external or internal disturbances, such as reaction-jet impulses, docking, impact, gravity torques, mass unbalance, crew motions, and rotating equipment. In an emergency, the reaction-jet system can be used to damp the station wobble also.

III. TECHNICAL APPROACH

Throughout this study program, S&ID has generally taken a conservative design approach for it is believed that a very high level of confidence will be required in the early multimanned space stations. Information on the approach employed in the overall space station design is contained in this section; for additional details on the approach to a particular technical area, reference should be made to the subsequent Technical Studies section.

While all activity under the study contract has been devoted primarily to the use of the space station as an orbital laboratory, there appears to exist (within both NASA and the Air Force) interest in the use of the space station for other applications. For example, the NASA Goddard Space Flight Center has been engaged in a number of activities that might lead to a manned astronomical observatory. The NASA Marshall Space Flight Center has been conducting (with industry participation) studies on the use of an orbital launch platform in conjunction with lunar and interplanetary missions. In addition, the Air Force has been investigating the feasibility of a so-called Military Test Space Station that could be utilized as a proving ground for specialized systems and operations unique to military requirements. Other potential applications can be foreseen in conjunction with a satellite communications network and as a military base of operations.

Studies under the NASA contract have maintained the basic concept of the self-deploying space station, i.e., a hexagonal configuration composed of a series of rigid modules which can be clustered into a compact arrangement for launch by a single booster. The initial configuration, illustrated in Figure 2, was composed of six rigid modules joined by inflatable material and arranged in a hexagonal shape that is rotated to provide artificial gravity. Three spokes joined the modules to a central hub which was to be used as a docking platform for two Apollo personnel transport vehicles. The three spokes were to be constructed of inflatable material which could be folded to achieve the compact launch configuration shown in Figure 3. One Apollo vehicle, launched with the space station, would serve as the means of escape for the three crew members in the event of booster malfunction and launch abort. When the orbit has been successfully established, the folded station is deployed automatically by mechanical actuators. The crew members remain in the Apollo until after deployment; however, after it has been fully developed and pressurized, they enter the station through the hub and begin the tasks associated with orbital operations. The present selected configuration retains the basic approach; however, a number of significant changes have been made where it was found that definite improvements in overall performance could be obtained.

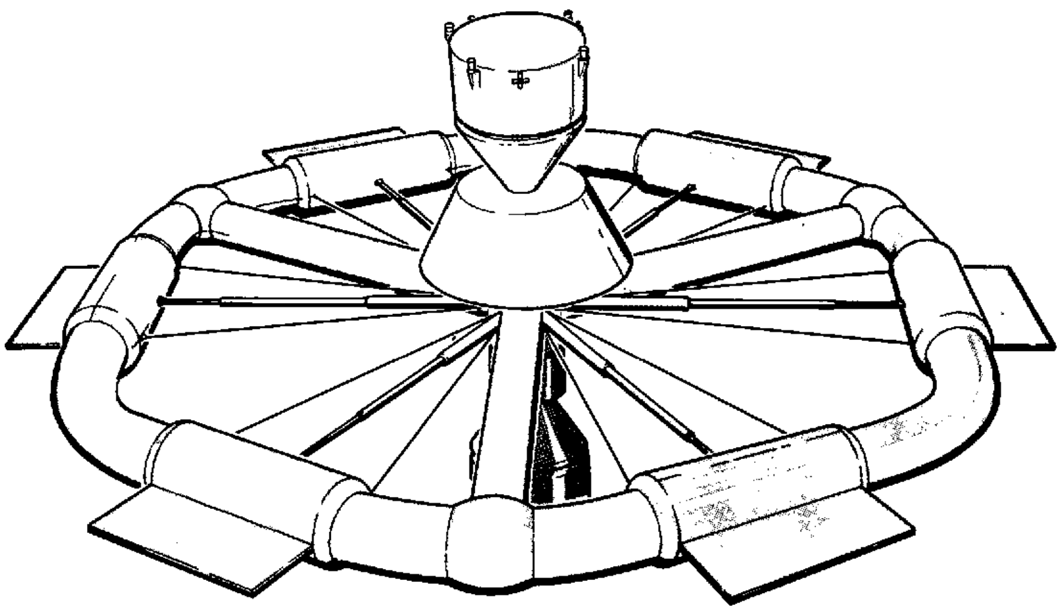
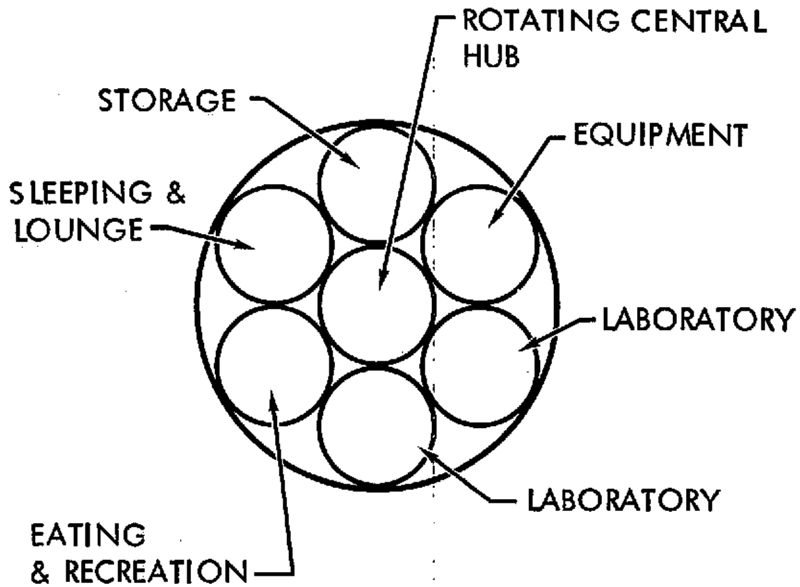


Figure 2. Initial Concept



SECTION A-A

INITIAL STATION LAUNCH WITH
PERSONNEL TRANSPORT VEHICLE,
ON SATURN C-5 (S-IB & S-II)

RESUPPLY & PERSONNEL TRANSFER
VIA SATURN C-1 OR TITAN III

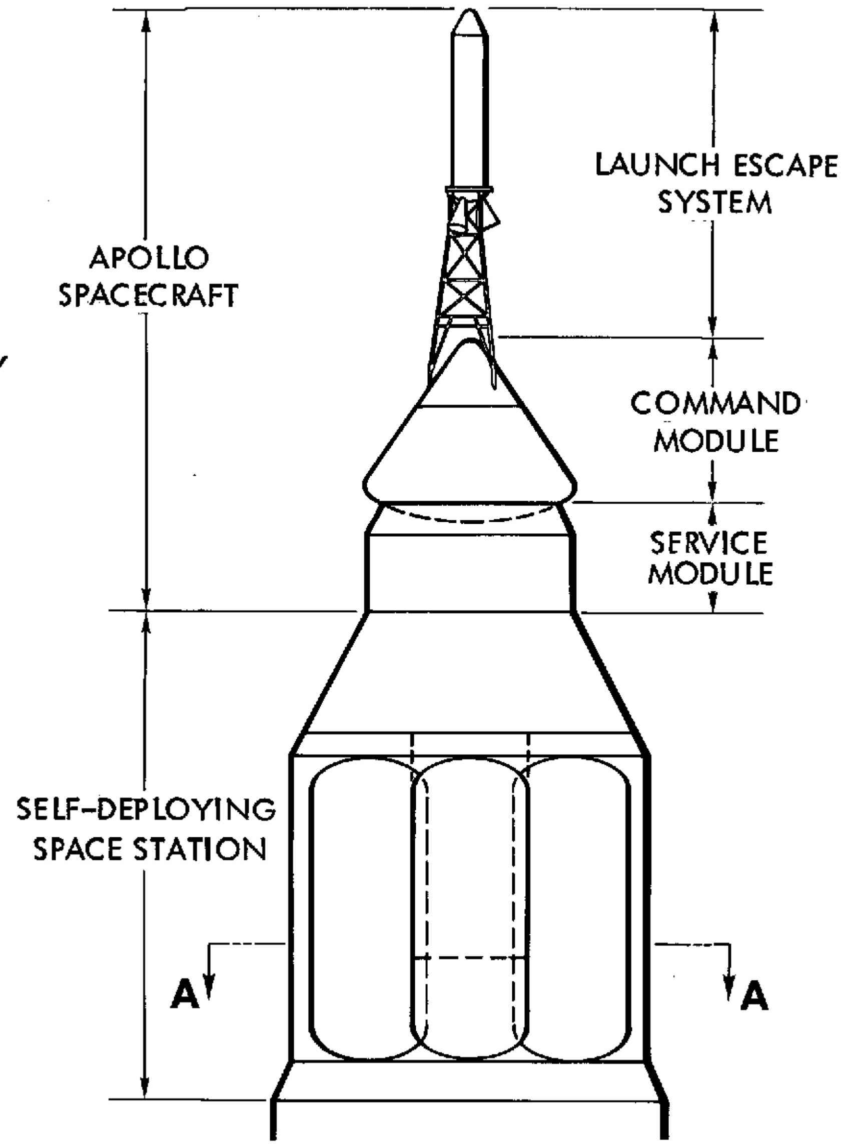


Figure 3. Launch Configuration

The 6-month study program was divided into three phases—Concept Evaluation, Initial System Analysis, and Final System Analysis—each of about 2 months duration. During the first phase, effort was concentrated on examining various space station configurations and developing requirements in all technical areas. In the second phase, various approaches to a system design were compared to determine an optimum approach. In the final phase of the study, the design details of the selected system were established to provide a better basis for confirming system feasibility and for examining development factors.

In consideration of the many unknown factors associated with operations in space, it has been generally concluded that it would be desirable to have the space laboratory available at the earliest possible date to explore many of the problems for which answers do not exist. Consequently, only presently planned launch vehicles and personnel transport vehicles were examined for use in conjunction with the space station. The only launch vehicle that would have the required order-of-magnitude payload capability is the Saturn C-5 (i.e., the configuration composed of the S-ID and S-II stages). Personnel transport vehicles could come from either the Mercury, Gemini, or Apollo programs. The Apollo was, however, chosen as the personnel transport vehicle because it enables three or more crew members to be launched and subsequently returned with each vehicle; the others carry less.

On the basis of the early launch possibility, a conservative approach has been taken in the space station design. In order to achieve a maximum level of confidence in the use of the station, only systems which have previously undergone extensive development have been recommended for use. A notable example of this appears in the choice of the power system for the space station. Of the several types of power systems that were studied, two types—the nuclear reactor (SNAP 8) and a solar dynamic power system—have certain features which make them look promising for use; however, neither will be fully developed before 1965. Solar-cell power systems, which have already been used in several satellite applications, have proven their reliability. Although a very large solar-cell array is required to generate the peak load required by the station, the reliability consideration is predominant and a silicon solar-cell system is considered most suitable for use by the earlier stations. After experience is obtained in the operation of the more advanced power systems, these could possibly be substituted for, or added to, the solar-cell system.

The predominant result of this conservative design approach is an increased confidence in the ability to achieve an operational space station at an early date. The limiting factor on the space station availability is undoubtedly the booster availability. The earliest possible date on which the space station could be launched appears to be in early 1966 when the first Saturn S-ID/S-II booster combination becomes available.

Because this space station will be the first large multiman vehicle to be operated in the space environment for extended periods of time, reliability of operation is a very important factor. The S&ID studies indicate that reliability of operation can be enhanced in a number of ways. In additon to the use of proven components and techniques previously discussed, it has been concluded that each of the seven major sections (i.e., the hub and six modules) of the space station should have independent environmental and power systems, so that a failure of either of these critical systems will not jeopardize the lives of the crew members or the success of the mission. In addition to its own environmental and power system, each module has an airlock at each end so that it can be isolated from the rest of the space station in the event of a major equipment failure or loss of pressure in that module. Similarly, each of the three spokes contains an airlock at each end, thereby dividing the space station into 10 separate compartments. A "shirt-sleeve" environment has been provided throughout the space station because it is not believed feasible to require the crew members to wear pressure suits, even deflated, on a continuous basis as do the crew members of Mercury, Gemini, and Apollo. Consequently, compartment airlocks would not normally be utilized except in emergencies when crew members would don pressure suits to enter a depressurized module to conduct repairs. While it is possible for the crew members to work in the interior of the space station in pressure suits, it is not considered feasible—in an early time period—for crew members to work on the exterior of the vehicle. Consequently, the space station has been designed so that there is no necessity for the crew to assemble elements in space or in any way assist in the deployment. Another important factor related to the reliability of operation is the stability of the vehicle. If the stabilization system should fail, it is exceedingly important that the vehicle motions do not become violent. To eliminate the possibility of such an event, the space station has been designed to concentrate the mass in the plane of the hexagon, thereby achieving a configuration whose moments of inertia represent that of an inherently stable disc.

While the initial space diameter was only 100 feet, it became apparent early in the study that a larger diameter vehicle would provide a more comfortable living environment for the crew members. With the NASA decision to develop the Saturn C-5 for the Apollo mission, it was possible to increase the diameter to 150 feet. The space station module diameter and length were established by the booster diameter and the launch configuration stability.

Cylindrical modules and curved modules (both of circular cross-section) were comparatively examined for possible application to the space station. The curved modules, when folded for launch, extend beyond the booster diameter and may create an instability during launch. Since they are also more difficult to manufacture, it was concluded that straight cylindrical modules represent the more desirable configuration.

Space stations crew sizes of from 6 to 27 were examined as variables in this study. Without an extensive task analysis for a specific mission, it is difficult to determine either the minimum number of men required to perform the basic functions or the number of men that can be utilized efficiently. Since time and the scope of effort did not allow such an analysis, the crew size was arbitrarily established to be 21, in order to provide a design point for sizing systems and determining weights and resupply requirements.

The Apollo is the only reentry vehicle planned for the 1966 time period that will be suitable for use as a personnel transport and resupply vehicle. Other vehicles have limited volume for resupply items and do not have air-locks to permit egress of the personnel in space. In this study, it has been assumed that the Apollo command module carries only three crew members; however, it is probable that the internal arrangement could be redesigned to accommodate a larger number.

The space station subsystems have been designed to function (with nominal maintenance) for an extended period in orbit. The maximum mission duration for the crew members in this time period has not been conclusively established. For the purpose of this study, the mission duration has been arbitrarily defined as 6 weeks for early space station operations. Operational costs have been examined as a function of crew size, mission duration, and type of reentry vehicle.

IV. TECHNICAL STUDIES

The technical studies which are discussed herein represent the results of the first 6 months of the feasibility study which was conducted by S&ID. Emphasis has been placed on presenting the results of the analyses conducted in the past 3 months; this includes a recommended technical approach, preliminary conclusions, and suggestions for future effort. The first 3 month's studies were presented in detail in the midterm report, SID 62-191, and are only briefly treated herein.

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CONFIGURATION ANALYSIS

The selected configuration for the self-deploying space station resulted from a comprehensive study of various vehicle configurations and design concepts and from investigations of critical technical problems within a given approach. A description of the recommended configuration and its characteristics are presented in this section. A discussion of the studies in areas of primary importance and a summary of several of the configurations considered—and from which the recommended configurations evolved—during the study are also presented.

The self-deploying space station, as described herein, is technically feasible and could be placed in full operational status by 1966, or earlier. The selected configuration represents an optimized overall arrangement. A compromise of individual subsystems is always required to produce an optimum integrated arrangement of all the required systems; and in order to achieve this total optimization, close coordination has been maintained among the various technical areas. Coordination between S&ID and LRC has permitted continuous mutual agreement on the major technical decisions which were made during the course of the study.

CONFIGURATION EVOLUTION

The selected space station configuration is the result of analysis of numerous arrangements and space station design concepts. A detailed discussion of these various arrangements was presented in the midterm report; however, a summary of the basic types of arrangements is presented here to indicate the evolution and refinement of design concepts reflected in the final "base-point" design.

Each candidate configuration served as a base-point design for investigation of the feasibility of design problem areas peculiar to that particular arrangement and as a basis for comparison of the various design approaches. Some of these design concepts were eliminated from further study as a result of these comparative evaluations. In some cases, subsequent arrangements were initiated to investigate an alternate approach to specific problem areas. As a result, the final configuration actually represents an evolution of the basic design concept that reflects refinements or changes of the design solutions of the various significant problems inherent in the self-deploying space station. The resulting final configuration is a composite of the most desirable features of each candidate configuration with the necessary compromises required to define an optimum total integrated system. A summary of the basic design concepts and space station configurations studied is represented pictorially in Figure 4.

Configuration I

Configuration I represents the original arrangement proposed by S&ID based on the space station concept developed by LRC. This arrangement was composed of six rigid cylindrical modules interconnected by inflatable material and arranged in a toroidal shape. Three inflatable spokes joined the torus to a central hub that was used as nonrotating control center and docking platform for two personnel-transport vehicles. The three spokes and the module interconnector sections were to be constructed of inflatable material which could be folded to achieve the compact launch arrangement. Deployment of the space station would be accomplished with actuators on each rigid module for extension of the modules and inflation of the fabric interconnections for rigidization of the deployed wheel. Flexible tension ties between the hub and modules would restrain the rigid modules in the deployed position. The rigid modules were 7 feet in diameter by 24 feet in length and located with a rotational radius of 50 feet. This original configuration was sized for earth launch by a Saturn C-3 booster.

Configuration II

The general arrangement of a 150-foot-diameter space station patterned after the original concept is shown as Configuration II. This arrangement differs from the original concept in several ways. The major differences are the station diameter, the location of the inflatable spokes, and the method of deployment. These changes were made possible by the added boost capability of the C-5 launch vehicle.

The diameter of the space station is increased from 100 feet to 150 feet. This increase in diameter of the rotating sections affords more desirable capability for artificial gravity simulation within the limits of human tolerance of rotational speed. In addition, the increased length of the rigid modules provides a significant increase in usable volume. Within the rigid modules, the floor was also changed to a curved section to maintain a more constant level of artificial gravity.

The location of the inflatable spokes was changed to permit attachment to the rigid modules rather than to the inflatable module interconnectors. This eliminates the complexity of a "T-joint" in the inflatable fabric and achieves a more efficient and simple method of construction for the inflatable structure. In addition to relocating the spokes, the amount of inflatable interconnecting fabric was minimized to provide maximum rigid module length. The rigid module length of this arrangement is 60 feet.

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The deployment concept for this arrangement considered use of the pressurization of the inflatable sections as the sole means of erection as an alternate design approach to the mechanical deployment method used in the previous study.

The three spokes that provide access between the living modules and the central lab in the hub are of flexible inflatable material. They and the module interconnectors would be evacuated as the payload configuration is assembled, and the spokes would be folded in such a way that a portion rests under two adjacent modules. At the time of deployment, pressure would be gradually introduced to the spokes to force all modules outward from the stowed position. As the shape of the modules approaches the wheel in form, the increasing pressure in the spokes would create tension in the spoke walls that should provide additional and final deployment motion. After deployment is complete, a release of pressure trapped in the rigid modules would be introduced to the connecting portion to attain rigidity.

Support of the space station during boost would be provided by a center interstage structure that interconnects the booster and the central hub. The clustered modules would be secured to the center interstage by straps. Release for deployment would be through explosive tension studs which free the straps.

Configuration III

Configuration III was investigated to determine the effects of various changes to the basic configuration. The major differences in this concept are the shape of the rigid modules, the consideration of a method for rigidizing the inflatable structure, and the investigation of a different method of booster attachment and boost load distribution.

The rigid modules are shown as curved sections to match the rotational radius and to maintain a constant level of artificial g at similar points throughout the module. The use of a curved floor in the module enables the crew members to feel a constant acceleration force as they move from one end of the module to the other.

Protection of the crew compartments and pressurized areas from meteoroid penetration was a primary concern. An investigation of a means of rigidizing the inflatable structure after deployment was conducted during the study of this configuration. One method which offered potential was the use of rigidizing panels which could be installed inside the fabric sections. These panels could be stored inside the rigid modules during launch and then used to line the inflatable sections after station deployment.

Station deployment utilized the flexible radial passages as pneumatic erection cylinders. Erection velocity would be regulated by ladder-type cables which are program controlled automatically during station deployment.

The booster-attach structure of this arrangement utilized the full diameter of the booster and clustered modules package for distribution of the boost loads into the payload. The boost load would be distributed from the booster upper stage into each of the clustered modules. Attachment of the modules to the booster stage and payload fairing and between the hinged modules would be accomplished with Marmon-type clamps. Release of these clamps would permit separation of the booster interstage fairing and free the modules from the payload to permit deployment.

Configuration IV

Configuration IV is similar to the previous arrangement; however, the extended length center section was eliminated and provision for multiple docking of personnel shuttle vehicles was considered.

Connection of the inflatable spokes to the module interconnectors, rather than to the rigid module, permitted elimination of the center section and provided protection for the inflatable spokes. Deployment of the modules by inflation of the spokes was assumed; however, the spoke attachment does not permit a straight line deployment of the spokes, which was possible with the previous arrangement.

Docking provisions for 10 Apollo vehicles was provided by two spheres, located above and below the central hub. Each sphere accommodated five personnel shuttle vehicles and permitted direct transfer from the shuttle to the central hub through individual airlocks.

Configuration V

The previous configuration employed rigidizing panels to provide added meteoroid protection. Configuration V represents an arrangement which provides maximum meteoroid protection in the rim of the space station. Elimination of the inflatable structure between modules resulted in a completely rigid structure for the outer rim. In addition, the zero-g laboratory section is located in the lower interstage structure. This permitted an efficient use of the interstage structure and resulted in improved boost stability of the payload by lowering its center of pressure and reducing the total payload length. Boost loads would be distributed through the interstage structure to a central support structure that supports the Apollo above the clustered rigid modules.

The primary features of this arrangement were the completely rigid rim, the use of hinges and actuators on the modules for deployment, and the optimum boost configuration. The rigid rim provided maximum protection and resulted in the requirement for inflatable fabric only in the radial spokes These spokes could be rigidized by the method discussed in the previous section if desired. The boost configuration of this arrangement is aerodynamically clean and is reduced in length by the height of the zero-g laboratory that is combined with the interstage fairing.

The three spokes would be manufactured from inflatable material and stowed folded within the confines of the cluster. They would be evacuated at the time of payload assembly so that no strain would occur as the payload experienced decreasing pressure in boost. Spokes do not assist in the deployment of modules, but would be drawn to the extended position by independent forces of deployment provided at each joint of the hinged wheel.

Configuration VI

The arrangement of Configuration VI was based on an all rigid wheel utilizing the deployment method of the previous arrangement but with curved modules and rigid sections in the spokes. This arrangement also accommodated multiple docking of Apollo shuttle vehicles.

The curved modules in the stowed position provided sufficient volume for stowage of short sections of rigid spokes between the modules and the central interstage section. The semirigid spokes are composed of two rigid sections connected by a short section of inflatable fabric. One rigid section is hinged to the center interstage section and the other to the rigid modules.

Multiple docking accommodations for Apollo vehicles were provided in the central hub section. Eight docking ports with individual airlock compartments were located around the periphery of the hub with one central docking port located on the central hub turning axis. All vehicles would be captured at the central dock and transferred by a motor and gear-driven respotting crane to the stowing docks on the periphery of the hub. The curved modules permitted a constant g level at the module floor and also provided a storage space for the rigid sections of spokes; however, the launch configuration resulting from the curved modules was undesirable.

Configuration VII

The desirability of an all-rigid structure space station for added meteoroid protection encouraged the evolution from the previous partially-rigid spoke arrangement to rigid spoke arrangement of Configuration VII. This configuration incorporated an all-rigid wheel and all-rigid spokes.

The complete mechanical hinging of the rigid spokes and rigid rim modules was accomplished by using a short hinged segment of rigid structure between modules to adapt to the motions involved. Hinges on each end of the radial spokes provided for the hinging of the components. An internal fabric liner at each hinged section provided a seal for the pressurization.

An alternate design concept for the multiple docking was also investigated in this configuration. This concept consisted of a docking sphere with a mounted central turntable. Eight pairs of guide rails radiated from the turntable and terminated at eight equally spaced docking ports around the sphere. A docking and transfer platform was attached to each set of rails. This platform transferred the captured Apollo vehicle from the central docking port to the stowage position at the terminal position of the radial tracks where an airlock for crew transfer was incorporated.

Configuration VIII

The configurations summarized in the preceding pages indicated a natural evolution of design concepts and desirable features. However, many of the configurations also represented specific desirable design characteristics. The final configuration, therefore, represents a composite arrangement that incorporates some of the specific desirable characteristics and reflects the natural evolution of design concepts. The resulting arrangement is presented as Configuration VIII. This arrangement provided the base-point design for the final selected configuration. The selected configuration is a refinement of this base point and was described in detail in a preceding section of the report.

AREAS OF SPECIAL INTEREST

Many of the design concepts and parameters investigated during the study are discussed in the previous section on configuration evolution. Some of the major areas of interest are presented in more detail in this section. The basic design assumptions, results of design comparisons, and advantages offered by the selected designs concepts are presented. The arrangement of the wheel and spokes in the stowed and deployed position are discussed, as well as the various methods of deployment. The configuration of the hub and the provisions for docking the Apollo shuttle vehicles are also discussed.

Module Design

During the course of the study period, S&ID has investigated the use of straight cylindrical modules as well as curved modules. The straight modules are advantageous in that they enable a very streamlined launch configuration to be obtained. These are also somewhat simpler to manufacture than curved modules. The primary disadvantages of the straight modules are that personnel will experience a change in gravity as they move through the module, and they will feel as if they were standing on an inclined floor when near the ends of the modules. The curved modules are advantageous since the personnel who will be housed in the modules are always at the same distance from the center of rotation of the space station; consequently, they will not experience any change in the level of gravity as they move from one portion of the module to another. Since the curved modules cannot be folded into a compact payload (i. e., a payload whose minimum cross section is no greater than that of the launch vehicle), it has been concluded that the straight cylindrical modules represent the more desirable configuration. In order to overcome the undesirable feature of the straight cylindrical modules, floors that simulate those in the curved module can be internally installed. In this way, the crew members would not experience any change in level of gravity as they move around within a localized area.

In most of the configuration studies conducted, the number of modules making up the space stage was limited to six. Although this number was selected in a rather arbitrary fashion, subsequent studies under the structural analysis area have indicated that this number appears to be optimum. The weight trade-off studies showed that both larger and smaller numbers of modules have resulted in a greater total space station weight.

The problem of sealing the joints between adjoining modules and between modules and spokes has been studied extensively. The joints must be made as nearly free from leakage as possible in order to minimize the amount of air which must be resupplied to the space station. Existing and planned vehicles such as Mercury, Gemini, and Apollo can tolerate relatively high unit leakage rates because their mission duration is short and their surface area is small in comparison to the space station. The aluminum wall structure, if manufactured under carefully controlled conditions, should have little or no leakage. It has been concluded that the well-designed O-ring seals, backed up by a metal bellows or sections of flexible material, would provide an adequate seal at the joints between the various sections of the space station.

Spoke Design

The spokes provide passageways for the crew members to move between the hub and the living and working modules in the rim of the space station. Spokes also fulfill the obvious function of maintaining the concentric location of the hub. Two spokes cannot maintain this location because of their inability to act in any direction but along their common center line. Three spokes are the minimum number that can establish and maintain a centralized location of the hub with respect to the modules. More than three spokes would, of course, increase the number of passageways from the hub to the modules; however, this does not appear necessary.

Spoke walls of the same multiwall structural configuration as the walls of the modules are necessary to provide equal meteoroid penetration resistance throughout the space station. Telescoping spokes would employ an O-ring seal that engages as the spoke approaches its maximum length. It is believed that a good O-ring seal design will prevent leakage around the telescoping joint. However, if leakage becomes excessive, the joint can be sealed by flexible tape which has a nongassing elastomeric adhesive.

Launch Configurations

In the early phases of the study, many of the space station configurations were shown with a central hub configuration that was considerably longer than any of the six modules. In these configurations, the modules were folded for launch against the central hub and were secured by straps. All of the boost loads were carried by the central hub including bending moments and thrust loads from the Apollo vehicle which was supported at the top. The central hub was also used as the interstage structure between the space station and the Saturn S-II booster. This particular configuration was attractive because it provided little or no interference with module deployment and because it contained a large internal volume. Stability analyses of this configuration showed, however, that the moment-of-inertia distribution more nearly resembled that of a sphere than of a disk. Since the moment-of-inertia distribution must resemble that of a disk in order to provide some measure of inherent stability, it was decided that a very short central hub was necessary. A weight analysis conducted in conjunction with the stability analysis indicated that a short hub could result in approximately a 1000-pound weight saving in the space station configuration, even though in using the short hub, it is necessary that the module cluster carry all loads which are encountered during the boost trajectory.

Deployment of Semirigid Stations

Space station configurations that use tubular flexible material for spokes and connections between rigid living modules are defined as semi-rigid arrangements. The deployed vehicle, once rigidized with the pressure of a habitable atmosphere and with a limited number of pre-arranged wire spokes, can present a space station within the regime of required stiffness. Additional contour definition can be provided if the nonrigid sections are backed-up by a metal mesh.

The flexible interconnectors may be bonded to metallic structure and should create no leakage problems at the joining ends. Deployment can be a straight forward application of force by extensible struts, which lock at the end of travel holding the living modules tight against the stabilizing wire spokes previously mentioned.

Liaison between LRC and S&ID at the outset of the study indicated that two problems existed in a station with flexible elements in the pressure structure: (1) Flexible casings, thin enough to be folded for a boost package would probably be too thin to satisfy the allowable leakage of atmosphere to the hard vacuum due to meteoroid penetration. Therefore, rigid external bumper panels over the flexible material would be required after deployment; or internal rigid panels must be installed after deployment and sealed to prevent leakage. (2) Flexible elements folded in a payload that must deploy in a hard vacuum must either be evacuated of all air prior to launch or they must be vented completely so that residual air does not expand inside the cavities during launch. If folds are evacuated on the pad, a small residual pressure in the confined payload could expand in unpredictable places, causing unsymmetrical or uncontrolled deployment. Consequently, after the modules are released from the cluster, collisions could occur or the inflatable sections might be severely damaged. Evacuation by venting in transit or by vacuum pump on the pad requires that flexible casings be free of closed folds; otherwise, the folds themselves might become sealed cavities in the presence of external pressure.

Deployment of Rigid Stations

Configurations of the all-rigid stations, described in the earlier midterm report, were based on design liaison meetings between LRC and S&ID. In this concept, the six modules stow in a manner similar to payloads previously discussed and the spokes stow in the center recess created by the modules. Deployment is all mechanical and is symmetrical.

Telescopic spokes have been used in the all-rigid space station configuration because the spoke length is greater in the deployed configuration than it is in the launch configuration. As can be noted from Figure 5, one of the earlier configurations, the spokes are attached at the midpoint of the module. Alternate configurations where the spokes attach near the ends of the modules and where the spokes do not change length have also been investigated. In these configurations, however, it has been found that deployment will cause the hub to rotate. It is also more difficult to provide the necessary attachments between the spoke and the module if the attachment is made near the end of the module.

A number of crude models of the space station configuration have been built, both by S&ID and by NASA. In the early investigations of the all-rigid configuration, a single hinge was used between each module. When the

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models were built and when the motions were mathematically analyzed, it was found that by using a single hinge between modules the configuration would not deploy without perceptible bind. Various corrective measures, such as the use of soft hinge-pin bushings, were investigated to determine some possible means of relieving this constraint.

A system of six modules and six hinges that has no binding in its motion has been devised. This hinge arrangement, composed of a double or "Vee" hinge pattern at each of the six joints of the wheel, puts each unfolding module in its own hinge mode by spanning the 60-degree joints with links. The hinge is referred to as the Buseman Hinge (after its creator, Dr. Adolph Buseman, of LRC) and is pictorially represented in Figure 6. Determination of deployment by the Buseman Hinge is considered fundamental in operation because each module rotates to deployment freely about its own two hinge center lines, which are parallel. Holding six modules in a wheel together by links as described would appear to have a certain looseness or flexibility like a necklace of tubular beads. However, the mechanized joint drives, which are required in order to deploy the station symmetrically, limit looseness to the accumulated tolerance on gear drives in the system. The double hinge joint with drive coordination was demonstrated by a Langley model wherein gear sectors from each side engaged and rotated smoothly and evenly about the connecting link.

The nature of mechanisms for symmetrical station deployment has been considered two ways: (1) an axial drive, incorporating motors phase locked to the power and driving epicyclic gear trains located cylindrically about each hinge center line (this method is presented pictorially in Figure 7), and (2) linear actuators incorporating motors phase locked to the power and driving screw jacks located on a like moment arm about each hinge center line (this method is shown in Figure 8). Both systems produce a coordinated moment about each hinge pin in about the same way and have built-in irreversibility so no motion occurs without power. Both can also operate forward and reverse to follow through on possible earth-bound deployment motion qualification with the system supported on a jury rig.

These methods of actuation were selected because (1) drive motors locked to the power cycle are very easy to coordinate without mechanical tie if they are not overloaded; (2) phase coordination of hydraulic actuators appears unreliable without precise control of the temperature of all the actuators; (3) cable systems need all the elements of the direct drives plus their own encumbrance; and (4) linkages between modules would be compounded because they could not swing in a single plane as all joints have two hinge planes 60 degrees apart.

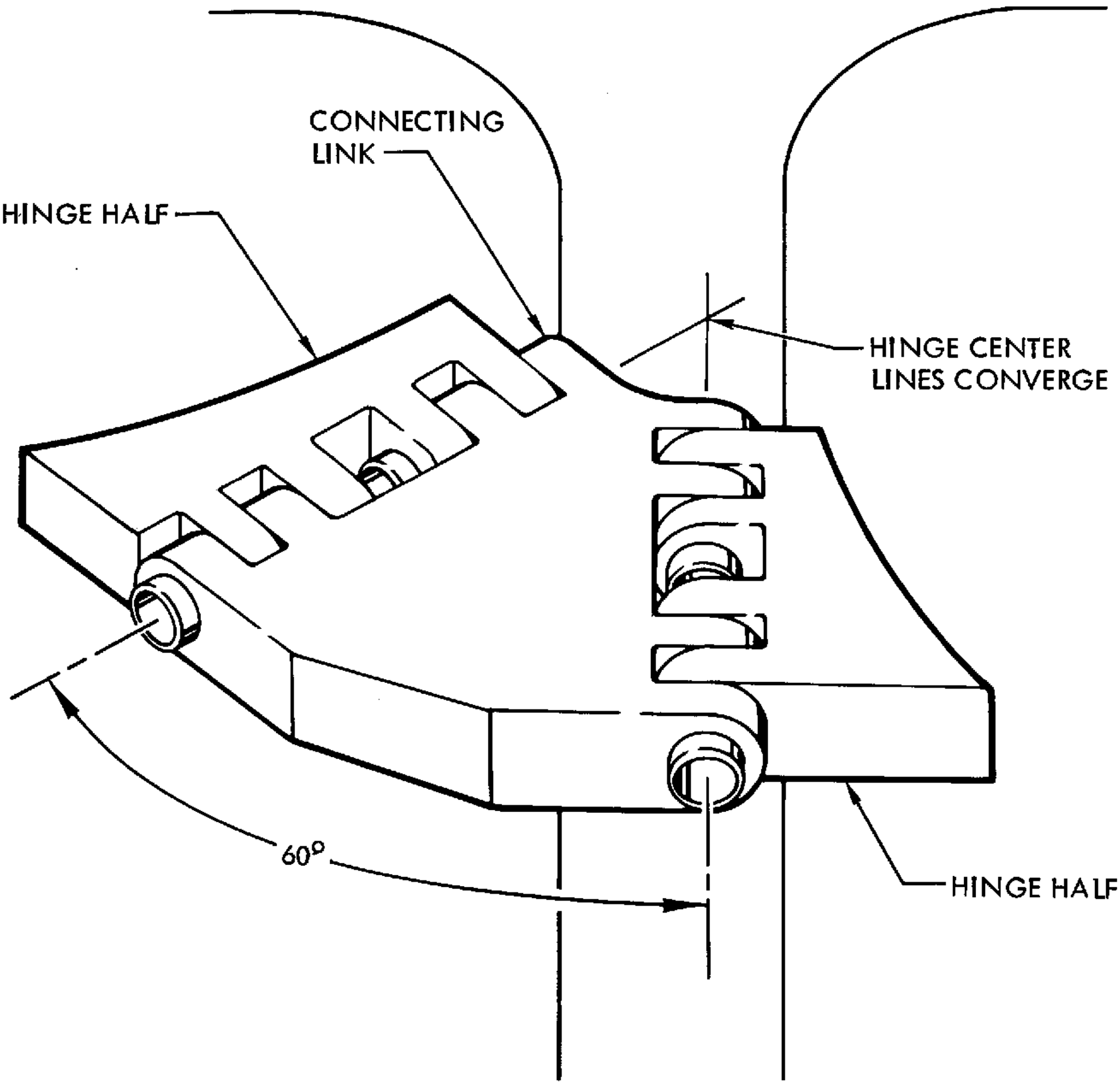

Figure 6. Buseman Hinge

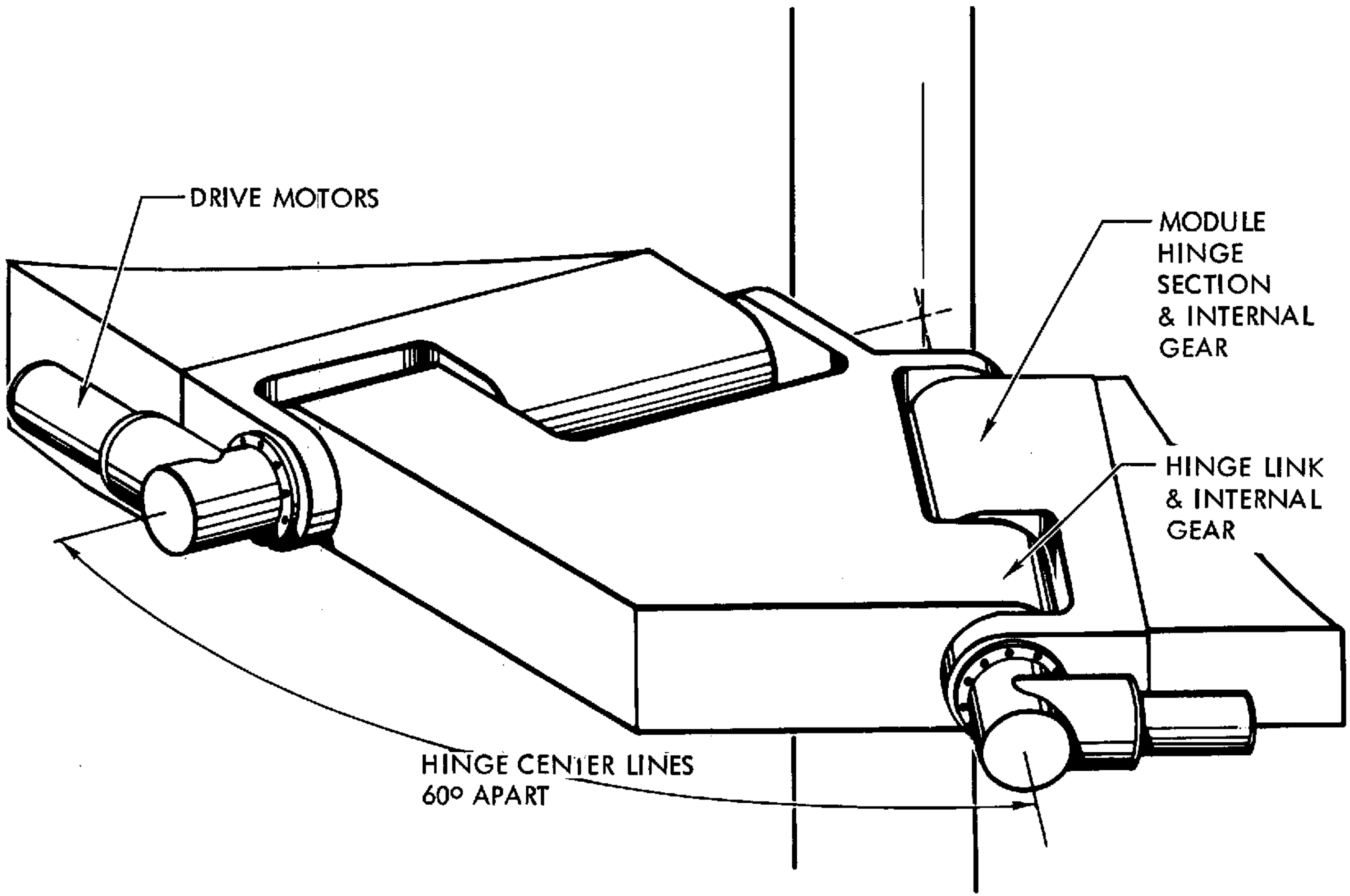

Figure 7. External Arrangement of Epicyclic Gearing on Buseman Hinge

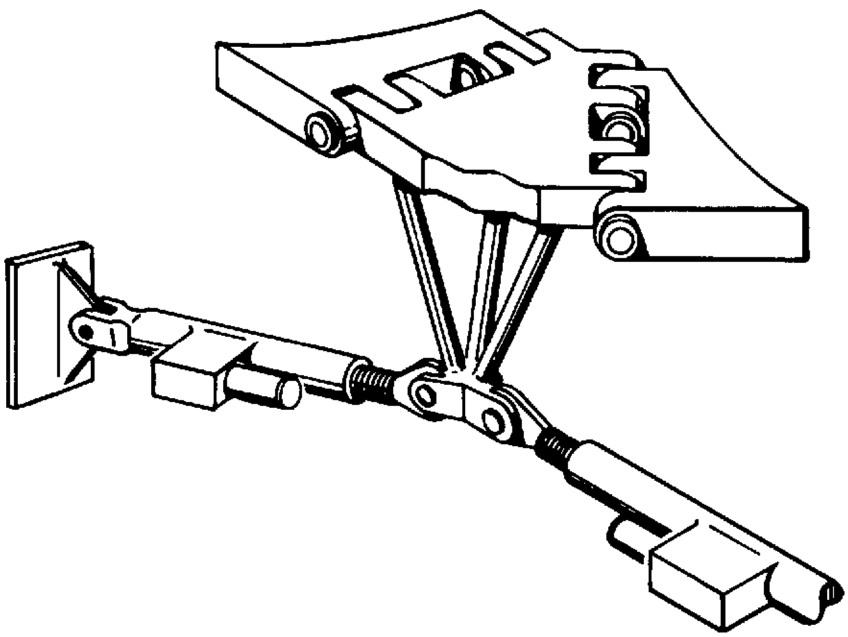

Figure 8. Linear Deployment Actuators on Buseman Hinge

Whether the telescoping spokes must be power driven to avoid high loads on the hinge-deploying system is not yet determined. Possible separation tumbling of the payload could require spoke assist through some portion of unfolding.

Scale models representing the geometry of the final study configuration, which are now being constructed, will be of assistance in establishing conclusions on the behavior of the deployment system and the motions of modules and spokes.

Hub Configuration Details

The 154-inch-diameter top of the central hub is the same as the mating portion of the Apollo vehicle which it supports in the boost configuration, and a 204-inch cylindrical section below it provides a reasonable clearance for six Apollo vehicles in the stowed position.

Six evenly spaced docks allow Apollo vehicles to be stowed opposite to each other to balance the station and to permit an independent entrance from three spokes into the interval between every other dock. Four docks on the 204-inch diameter is no problem; however, four docks on a smaller diameter, which provides minimum clearance for airlock operations, appear to compromise the spoke entrances and the available volume for the scientific laboratory below.

An early configuration, Figure 5, which shows stowing docks on a rotating turret included an array of rollers around the outside of the hub compartment at the plane of the floor. In the final configuration, turret rollers have been used on the upper skirt extension of the hub compartment. This removes them from the external wall of the pressure vessel and directs the launch loads of the Apollo system axially down the wall of the hub without an eccentric load path. The turret has been bearing mounted directly to the upper central airlock because this additional constraint to rotation of the turret localizes the concentricity problem associated with airlock to Apollo cavity integration.

Payload Cluster Details

Support of the central hub of the station during boost is provided by the upper ends of the stacked modules. A launch configuration with internally stowed rigid spokes precludes the use of a hub structure extending the full length of the modules since the spokes themselves occupy the cavity formed by the module cluster. The hub supports are extensions of the compartment wall which protrudes downward to join fittings located on each of the modules.

Release for deployment can be the boost separation signal which detonates explosive studs holding the cluster together.

Clamp fittings at the top of the modules do not hamper the deployment motion after they are released because they do not lie in the normal path of the motions. Suitable measures must be taken so that no explosive damage occurs to the station by detonating the studs.

Support of the payload on the boost vehicle appears to lend itself most readily to a system of fittings around the lower periphery of the living modules, which upon separation permit the payload adapter to pull away with the booster. This leaves no encumbrance to deployment. Figure 9 shows this type of support. Hold-down studs can align with the direction of separation, therefore, performing only under tension with shouldered interfaces on the fittings to resist the side load. A support system of the nature described above can be completely unsecured for deployment with explosive studs whose only load is tension. The lower connections to each module requires three studs, and each of six joints at the top requires one. This is a total of 24 studs that free the system.

Apollo Stowing Accommodations

The single Apollo vehicle is an integrated part of the space station boosted payload. After booster separation, the Apollo will detach itself, turn end-for-end, and engage a personnel airlock dock located directly under its boost position on the station turning axis. Before personnel can enter, however, a balance of pressure between the station and the Apollo command module will be necessary. This is a simple operation involving the use of bleed valves in the airlock doors.

The most ideal docking position on the rotating space station is on the axis of the hub. Since there are to be a number of vehicles docked at the space station at any given time, it is apparent that not all can occupy the position on the axis of rotation. When the Apollo vehicles are located elsewhere, care must be taken to assure that they are placed symmetrically to avoid any shift of the spin axis. Similarly, when the vehicles are being moved from the docking position to the stowage position, a means must be devised to prevent any mass unbalance that would create a wobble in the space station.

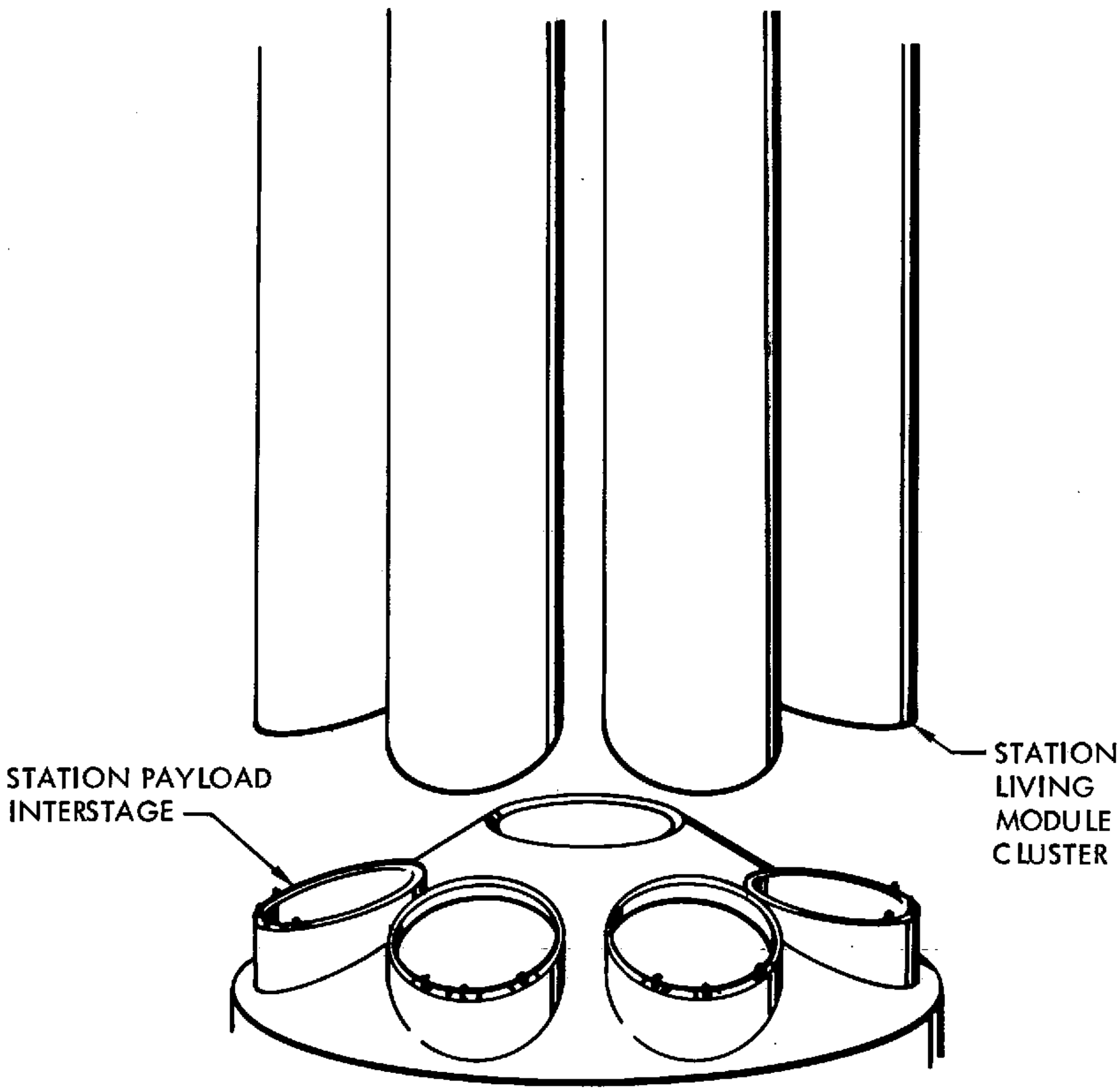

Figure 9. Payload Separation

A number of techniques to accomplish the multiple vehicle docking were investigated. The only concept that was found to be without a major problem area is the one illustrated in Figure 5. Docking and stowing are accomplished in the following manner. The entire upper section (turret) of the hub is mechanically driven in a direction opposite to the space station rotation, so that it becomes essentially nonrotating in inertial space. An electric drive mechanism brings a stowing boom and attachment ring over the axis of rotation. The incoming Apollo vehicle engages the attachment ring; and, if desired, an attachment can be made to the central airlock in the hub to enable the crew members to exit from the Apollo command module at that time. After the personnel exit, the boom moves the Apollo over to one of the circumferentially located stowage positions on the hub. Power to the mechanical drive is shut off, and friction in the bearings allows the turret to approach the space station rotational velocity. It will be indexed to stop at a predetermined position where a tubular passage is extended by the force of the space station internal pressure to engage the Apollo airlock. Thereafter, crew members may enter the stowed Apollo vehicles for transfer of equipment, and supplies. When not in use, the booms and attachment rings stow in recesses around the turret of the hub.

Each time an Apollo vehicle docks at the space station, all of the tubular passageways must be retracted to permit counter rotating of the turret. These are retracted by hydraulic actuators and are locked in position. Although an Apollo vehicle cannot dock without the counter rotating of the turret, it does appear that the vehicles could leave the dock in an emergency without the counter rotating of the turret. In leaving the space station, there is little risk of collision with the spokes or modules.

The normal departure from the station would involve counter-rotating the turret, moving the departing vehicle to the central docking location, and, after departure, rearranging the Apollo vehicles in a symmetrical pattern. After the Apollo vehicles have been relocated, the turret will again be permitted to approach the station rotational velocity. It is not believed feasible to continuously counter rotate the turret because of the power consumption involved and the difficulty in lubricating the bearings which are exposed to the vacuum.

FINAL CONFIGURATION DESCRIPTION

The general arrangement and major characteristics of the recommended space station configuration are presented pictorially in Figure 10. The selected configuration is a hexagon-shaped vehicle composed of six rigid modules connected to a central hub by three rigid telescoping spokes. The rigid modules are mechanically hinged to eliminate nonrigid interconnectors. The maximum diameter of the space station in the deployed configuration is 150 feet across the corners with a central hub height of approximately 29 feet. The launch configuration, with the modules retracted, is approximately 33 feet in diameter with a height of approximately 103 feet, not including the Apollo shuttle vehicle.

The space station was designed to simultaneously accommodate seven Apollo personnel transport/resupply vehicles which, based on three men per vehicle, represents a nominal crew size of 21 men. The available volume (and functional subsystems) within the space station is sufficient, however, to accommodate larger crews sizes for extended mission durations if desired. Docking facilities, located around the periphery of the central hub provide storage space for the Apollo vehicles and permit simultaneous emergency escape of the entire crew at any time.

A single Apollo vehicle and its crew could be launched with the self-deploying space station. The total launch weight is 170,300 pounds, including the Apollo Spacecraft, which is well within the capability of a single Saturn C-5 booster vehicle. A brief summary of the space station weights is presented in Table 1; a more detailed discussion of the weights is presented in a subsequent section.

Launch Configuration

All of the major structural elements that make up the station—i.e., hub, spokes, and modules —are mechanically folded into a package that forms a payload suitable for boost by the Saturn C-5 launch vehicle. The hub section of the station is secured to the top of the modules, which are radially clustered in a vertical stack for the boost. The cavity formed at the center of the cluster of modules is used for stowage of the three spokes which telescope to be compatible with the launch configuration. The spokes do not contribute to the structural stiffness of the payload. They are connected to the hub and modules prior to launch in a manner that will permit the configuration to deploy freely after it has been placed in orbit. The clustered modules are stowed below the boost flare, which is a part of the hub, and so remain after deployment. The primary purpose of the boost flare section is to provide protection for the modules during boost. The crewmen accompanying the space station during launch are located at the top of the payload in the Apollo vehicle. The Apollo vehicle provides the

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required capability for escape of the crew in event of failure during launch and is also the means for returning the crew to earth at the end of the mission. The total payload rests on top of the final boost stage on an adapter that is permanently attached to the booster. Separation fittings between the adapter and the bottom of the clustered modules tie the bottom of the cluster together to complete the structural stack.

Table 1. Weight Summary

Item	Weight (lb)
Structure	75,150
Equipment	63,450
Apollo Vehicle (3-man crew)	15,900
Apollo Launch Escape System	5,800
Interstage Structure	10,000
Launch Configuration	170,300
Less Interstage and Escape System	15,800
1 Apollo (3 men)	154,500
6 Apollo Vehicles (18 men)	95,400
Orbital Configuration (Full Crew 21 men)	249,900

Deployed Configuration

The hub, spokes, and modules of the station are hinged together in the launch configuration. Fittings at the top and bottom of the modules free both the hub and the payload adapter when booster separation occurs to permit deployment immediately after the orbit has been established. Deployment occurs by actuators in a symmetrical manner so that collisions between components of the station are not possible. Near the conclusion of deployment, "finder latches" at each hinge joint are extended to capture pins on the adjoining modules and are then drawn tight, to lock the joints and provide force for effective seal against air leakage of the joint.

The deployed arrangement is a hexagonal shape composed of six 10-foot-diameter modules connected to the hub by three rigid spokes. The modules are approximately 75 feet long and the 5-foot diameter spokes are approximately 48 feet long in their extended position. When deployment is completed, all modules and spokes lie in the same plane. Spoke lengths are not planned to be precisely controlled during the original extension because a spoke which does not extend to a sufficient length could act as a restraining tie rod and prevent completion of the space station deployment. Since the spokes control the proximity of the hub to the true center of the space station, it is believed that a final adjustment of the spoke length should be accurately performed internally by the crew after the station is locked in its deployed configuration.

The walls of all the modules and spokes are of the same structural configuration, i.e., multiwall honeycomb and insulation so that all exposed surfaces are equally resistant to meteroid penetration. The exterior surfaces will be aluminum alloy; the side directed away from the sun will be cleaned and polished; the side facing the sun will have a black strip mosaic finish.

Large panels of solar cells (320 square feet) are attached to each module and will unfold upon deployment completion. Each panel provides sufficient power to operate the equipment in the module to which it is attached. Solar cells for the hub are mounted on the lower side of the boost flare. It is, of course, necessary that the rotating space station, with its fixed solar-cell arrays, be oriented so that the spin axis is pointed toward the sun. The orientation is such that the docking facility on the hub is directed away from the sun.

Airlocks have been placed at each end of each module and at the ends of the spokes so that any of the 10 compartmented areas so formed may be sealed off from the others in the event of critical equipment failure or extensive damage in the area. Through use of the airlocks, crew members wearing pressure suits will be able to enter the isolated areas to conduct repairs or, for the central hub, to simply pass through it to reach the Apollo vehicles to return to earth.

INTERNAL ARRANGEMENT

The interior has been arranged to utilize the station configuration for maximum privacy, comfort, and convenience during extended orbital duty. A general arrangement of the space station interior is presented in Figure 11. The central hub contains docking, instrumentation and controls, and a zero-gravity laboratory. The six modules of the space station contain living quarters, crew work areas, and laboratory facilities. To avoid jeopardy to the crew members in event of equipment failure, the hub and each of the six modules contain individual, self-sufficient environmental and power systems. The environmental system is based on an oxygen regeneration cycle; however, stored liquid oxygen is provided as a backup to the regeneration unit. In

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addition, sufficient oxygen and nitrogen are stored to completely repressurize the space station two times. Consequently, a significant amount of the volume in the modules and hub is occupied by equipment associated with the environmental system.

Module Arrangement

In the modules, equipment is arranged on both sides of a 32-inch aisle. Consoles and displays requiring a seated operator are recessed in the wall to avoid obstruction of the aisle. Recent information indicates that the placement of seated crewmen facing the direction of rotation will minimize the psychological and physiological effects attributed to the rotation.

With the utilization of cylindrical modules in a hexagon configuration, it becomes necessary to add a secondary floor in the module to prevent excessive deviation of the gravity vector from a position normal to the floor when the station is rotating. Progressing from the longitudinal center of each module, the basic floor was developed by a series of arcs of increasing radii about the space station center of rotation. The arcs were then reduced to straight floor segments reasonably normal to the turning center and connected by stairways to facilitate fabrication. This change in radius of rotation will produce a gravity variant from the longitudinal center to the end of each module of approximately 0.20 g, a differential which is believed to be acceptable by the crew members.

Fuel tanks for the attitude control system are stored internally as close to the control jets as possible and yet remain consistent with an optimum living and working arrangement.

Access to the internal module wall to facilitate repair in the event of meteroid damage is a necessity. This may be accomplished either by hinging the modular packaged equipment on the aisle at the floor line and rotating the top of the equipment into the aisle to permit access to the wall, or by installing the equipment modules on drawer slides to permit pulling the entire rack into the aisle.

Equipment rack depth has been held to a maximum of 24 inches and a height of 72 inches to simplify handling and access. Internal wall access in the aisle area is accomplished by removable floor panels.

The internal arrangement is based upon a maximum complement of 21 crew members working a 3-shift 24 hour day. For extended periods, the most desirable arrangement appears to be separation of the three 7-man crews. This permits each crew a duty cycle undisturbed by those on another schedule. The basic hexagonal configuration of the station simplified this arrangement. Isolation is achieved by selecting three alternate modules for living quarters separated by work modules. Modules having the radial spokes attached to the hub have been selected as work modules. This permits a duty crew access to all work modules and to the hub without the necessity of entering a living area.

Living Modules

The three living modules are identical and completely self-sufficient for a crew of seven. Basically, the living module confines sleeping, dressing, and hygienic functions to one end and food preparation and consumption and recreation at the other. Sleeping is provided for six of the crewmen at one extreme end of the living module for maximum separation from the centrally located electronic and mechanical equipment. Bunks in the sleeping compartment are arranged with three on each side of the central aisle. The seventh crewman has a bunk just outside the sleeping compartment. Across the aisle from this bunk is the module control center. The console contains intercom, environmental monitoring instrumentation, and a station alarm system. The primary function of the stand-by crewman will be to monitor environmental conditions and alert the crew in the event of an emergency.

A shower and personal hygiene area with two basins, two mirrors, and storage for electric razors, toiletries, and other gear is provided. Water used by the shower and hygiene area is purified by the water reclamation system. To minimize plumbing, the shower and hygiene area are adjacent to each other. A clothing storage locker, dressing area, and a small laundry are located across the aisle from the shower.

To maintain the self-sufficiency of each module for a short period of time, a smaller (but adequate) food preparation area has been included in each module. Likewise, a food storage area containing all food required for the module is provided. The effects of an emergency are reduced by including a 2-day surplus in each of the three living modules. A crew displaced by an emergency in a living module would have a food supply for 4 days, which should be an adequate period for repairing the module. The eating and recreation area occupies the remaining length of the living module. The living module contains ample volume for storage of space suits and other equipment.

Work Modules

All laboratory material and testing equipment has been arranged in the three work modules. The equipment associated with testing of a specific subsystem (e.g., propulsion, space environment, etc.) has been arranged in the same module whenever possible. However, with the requirement of weight distribution for station balance, it has been necessary to divide the laboratory equipment of some subsystems between two modules. This division was made as an arbitrary volume allotment and does not necessarily constitute the final arrangement. However, at the present time, the sepation imposes no apparent problem on the laboratory procedure.

One of the work modules contains the station command center which includes primary station communication and control equipment. The laboratory equipment is divided among the work modules to give an equal total equipment volume.

A sick bay and dispensary area is located at one end of one work module. Three bunks are arranged on one side of the central aisle with a fourth across the aisle at working height. The fourth bunk serves as an examining table or, in an emergency, as an operating table. Adequate volume is available above the fourth bunk for the dispensary.

The remaining volume at the ends of the work modules may be used for additional living area, maintenance and repair facilities, or other miscellaneous activity.

Central Hub Arrangement

A primary consideration in the interior arrangement of the central hub was to provide isolation of the zero-g laboratory from personnel traffic and also to provide maximum comfort and safety for the crew. An arrangement incorporating two separate compartments within the central hub was selected to satisfy this consideration. An upper compartment provides for crew transfer to the spokes, the zero-g laboratory, or the Apollo vehicles which are docked around the periphery of the hub. The lower compartment provides space for conducting zero-g experiments.

The central hub section is rigidly attached to the spokes and rotates with the space station. A zero-g laboratory compartment is located within the rotating hub on a motor-driven platform. The platform is rotated in the opposite direction from the space station at the correct speed to maintain a relative rotation speed of zero. The rotating platform is enclosed with screens to eliminate adverse psychological effects on participants when viewing the rotating walls of the hub. The available area for experiments is approximately 45 feet square. Entrance to the compartment is through a hatch at the top. Ample storage space is available in the compartment for personnel engaged in experimental programs of extended duration.

The upper transfer compartment provides access to all the spokes and to the Apollo vehicles. Access to the spokes is by three tunnel passageways located around the nonrotating sections. Access to the docked Apollo vehicles is by individual airlocks located around the periphery of the upper section.

The structural configuration of the central hub is the same as that of the modules. Load path connections to the hub are at frames located on the external skirt. This eliminates connecting structures crossing the header joints of the pressurized compartment and permits design of the headers for pressure loads only.

The flared skirt, attached to the lower section of the central hub, is primarily required as an aerodynamic fairing during boost. It remains with the hub after deployment and provides an additional layer of material for meteroid protection. It also provides an efficient mounting platform for solar cell panels to provide power for the central hub. The bottom of the flared skirt will accommodate 312 square feet of solar arrays in three panels between the spokes. The location of power supplies on the hub eliminates the need of power transmission from the solar panels on the rotating modules.

STRUCTURES AND MATERIALS

The structures and materials work performed during the study has been aimed at establishing pertinent structural design criteria, selecting structural materials and a type of construction, and supporting overall configuration and weight analyses. The final configuration analysis considers the internal load distribution to the space station components and a detail sizing of the structural members. It identifies problem areas and weight optimization studies that will be required to arrive at a minimum structural weight vehicle compatible with "normal" design optimization procedures and with the "conservative" design approach that has been used to establish the space station characteristics.

STRUCTURAL DESIGN CRITERIA

The structural design criteria chosen for use in this study is divided into two environmental regimes. The first regime is the internal pressure and the loading environment, consisting of wind and inertia loads encountered on the ground, during boost and trajectory, and in orbital flight. The loads encountered during deployment are considered to have only local effects and for this reason are not included in the study. This analysis is required when the kinematics of the deployment system are defined.

An ultimate factor of safety of 1.5 (limit load multiplied by 1.5) is used in all cases, except for cabin pressure. To provide maximum safety of the crew, the ultimate factor of safety for cabin pressure is 2.0.

Limit loads were developed for the following conditions:

Ground

Condition 1, Prelaunch: Booster and vehicle erected to a launch attitude and exposed to a wind gust of 72 mph,

$q = 13.1$ psf,

$N_x = 2.0$ g

Trajectory

Condition 2, Maximum dynamic pressure: (Identified by a

$q\alpha = 120\ \text{lb/ft}^2$ - radians),

$N_x = 1.83$ g,

$N_y = 0.5$ g

Condition 3, Maximum Wind:

$q = 122$ psf,

$N_x = 1.5$ g,

$N_y = 0.5$ g

Condition 4, First stage burnout:

$N_x = 4.15$ g,

$N_y = 0.5$ g

Orbital Flight of One Year

Condition 5, Cabin pressure:

$p = 10$ psi,

$N_x = 0$ (normal to station plane)

$N_y = 1$ g (normal to spin axis)

Condition 6, Deployed, Spin-Up Accelerations:

$\dot{\omega} = 0.00056\ \text{radians/sec}^2$ producing a tangential force of 0.0013 g at the rim

Condition 7, Deployed, Pitching Accelerations:

$\omega = 0.0062\ \text{radians/sec}^2$ (about any axis normal to the spin axis) producing a normal force on the module parallel to the spin axis of 0.0144 g

Boost trajectories for the Saturn C-5 configuration with the space station payload is expected to impose moderately high heating rates and surface temperatures which are dependent, in general terms, on the following:

1. The flight trajectory describing the relationship between velocity, altitude, and time

2. The body configuration; i.e. the physical shape, type or types of material, thickness, and thermal properties of the material

3. The availability of a heat sink to which the thermal energy derived from aerodynamic heating is transferred

Temperature data obtained from the Saturn S-II data indicate that an operating temperature of 350 F for a period of 5 minutes during boost is reasonable for this study.

The second environmental regime chosen as a part of the structural design criteria is that of space where the effects of meteoroid penetration, radiation, and vacuum and associated temperatures on the structure are considered. These factors (i.e., the meteoroid flux and penetration effects, radiation, temperatures, etc.) are covered further in other sections of this report.

MATERIAL SELECTION

The considerations in the selection of a material for use in a space station are strength/density ratio, fabrication feasibility, creep resistance, fatigue resistance, meteoroid protection, resistance, and the effect of vacuum environment.

The strength/density ratio in the low-temperature region (i.e., up to 350 F) was found to be highest for alloys of beryllium, high-strength steel, titanium, aluminum, and filament wound fiberglass. Work performed early in the study indicated that the weight of structure required to withstand the module loading conditions had assumed secondary importance in view of the weight required in a meteoroid bumper to protect the crew and equipment from meteoroid penetration. For this reason, and joining considerations, the filament wound fiberglass material was discontinued. Of the four alloys, aluminum is the best structural material from the aspect of secondary radiation. Beryllium has a very high secondary emissivity cross section for proton and helium induced reactions that emit neutrons. Since neutrons contribute more to the total dose than any other type of radiation, the neutron flux should be kept as low as possible. Little is known concerning the secondary emissivity cross section of titanium and steel. However, a fair approximation of this cross section for high energy protection is the area of the nucleus. The area given by the formula $\delta = \pi(1.1 \times 10^{-13} A^{1/3})^2$, where A is the atomic weight of the material, shows that the higher the atomic weight of the material the higher the cross section. Thus titanium with A = 47.9 will give a lower cross section than iron (high-strength steel) with A = 55.85. Thus, it is possible to conclude that titanium would be better than high-strength steel. The cross section data is know for aluminum and it is lower than that for the other materials.

Aluminum then seems to be the best structural material from the standpoint of radiation since its mechanical properties are not appreciably affected and its secondary emissivity cross section is small. Little is known at this time about the effect of very high-energy particles on structural metals.

Considering these factors, i.e., the high strength/density ratio in the temperature range considered and the ease of fabrication, it was decided to use an aluminum alloy. Since one of the main functions of the structure is to maintain cabin pressure (this can be best accomplished with welded joints), 2014-T6 and 6061-T6 aluminum alloys were considered. The 2014-T6 was chosen because of the better strength/density ratio and the favorable experience in welding this alloy during manufacturing. It is low in cost and relatively easy to fabricate. The external skin temperature was assumed to be 350 F for a period of 5 minutes due to aerodynamic heating during trajectory. The temperature of the space station interior wall structure, which is enclosed in the meteroid bumper structure, is expected to be maintained at 70 F while the station is in orbit.

Creep and creep rupture phenomenon are not expected to be design parameters, since high loads at high temperature occur only for a short period of time. Fatigue is not expected to be a design criterion, since stress levels are relatively low due to other design considerations.

The use of inflatable materials on the vehicle will be held to a minimum due to its low resistance to meteoroid penetration and possible degradation of material properties in a vacuum. Astro Research Corporation performed a study on inflatable materials for use in the self-deploying space station, and the results are presented in Appendix B. A self-sealing wall construction for use in either flexible or rigid module construction was investigated. The concept tested employs several layers of alternating nonreacted resins and catalysts capable of quickly reacting into a flexible, elastomeric product upon contact with each other. The various ingredients are separated by thin plastic film membranes. Upon penetration, local mixing takes place and the resulting reaction provides a clot, which stops subsequent leakage. This concept could be developed and used to seal punctures up to 1/16 inch in diameter in the inner structural wall for a slight increase in module weight. It could eliminate or reduce the requirements for a leak detection system.

The problems with inorganic materials in outer space are the evaporation or sublimation into the space vacuum and the possibility of a decrease in mechanical properties. The sublimation of aluminum is only 4 by 10^{-6} inches per year at a temperature of 1020 F. This is an insignificant loss of material and will be orders of magnitude less at the structural temperature of the space station in orbit. It will occur first on the meteoroid bumper and second on the outer sheet of the structural wall. If the outer sheet of the structural wall is penetrated, this loss of material could occur from the honeycomb core and inner face sheet without a significant loss in strength.

There seems to be no good evidence of any important decrease in mechanical properties of inorganic material in vacuum, as compared to their properties under the atmosphere in which they are ordinarily used, provided the temperature is not so high that appreciable sublimation occurs (Reference 1). In fact, the properties are usually better in vacuum because of the freedom from corrosion.

CONFIGURATION ANALYSIS

The configuration analysis performed during the study consisted of a central hub analysis, a meteoroid bumper design, a comparison of four types of construction, an analysis to determine the optimum number of modules, and a thermal stress analysis for the station in the space environment. The central hub analysis was conducted using four early configurations having the inflatable material in the spokes. The results of this investigation were presented in SID 62-191 (Reference 2) and are not repeated here because they are no longer applicable. The results of the remaining analysis follow.

Meteoroid Bumper and Module Wall Design

The midterm progress report presented a meteoroid bumper and module wall design of sandwich construction. It consists of four 0.030-inch thick sheets of aluminum spaced 1 inch apart as shown in Figure 12. The first space, from the inside out, contained aluminum honeycomb material and when bonded to the face sheets constitutes the primary load carrying structure. The two remaining spaces were filled with glass wool and, in conjunction with the other two aluminum sheets, performed the function of a meteoroid bumper. Honeycomb was chosen as a core material to provide rigidity to the wall structure and resistance against crack propagation should the face sheet be penetrated by meteoroids. The penetration of a single and only skin of the skin and longeron type construction could mean the loss of an entire structural panel due to tearing of the sheet under load once it had been penetrated. Skins supported by honeycomb and operating at a relative low

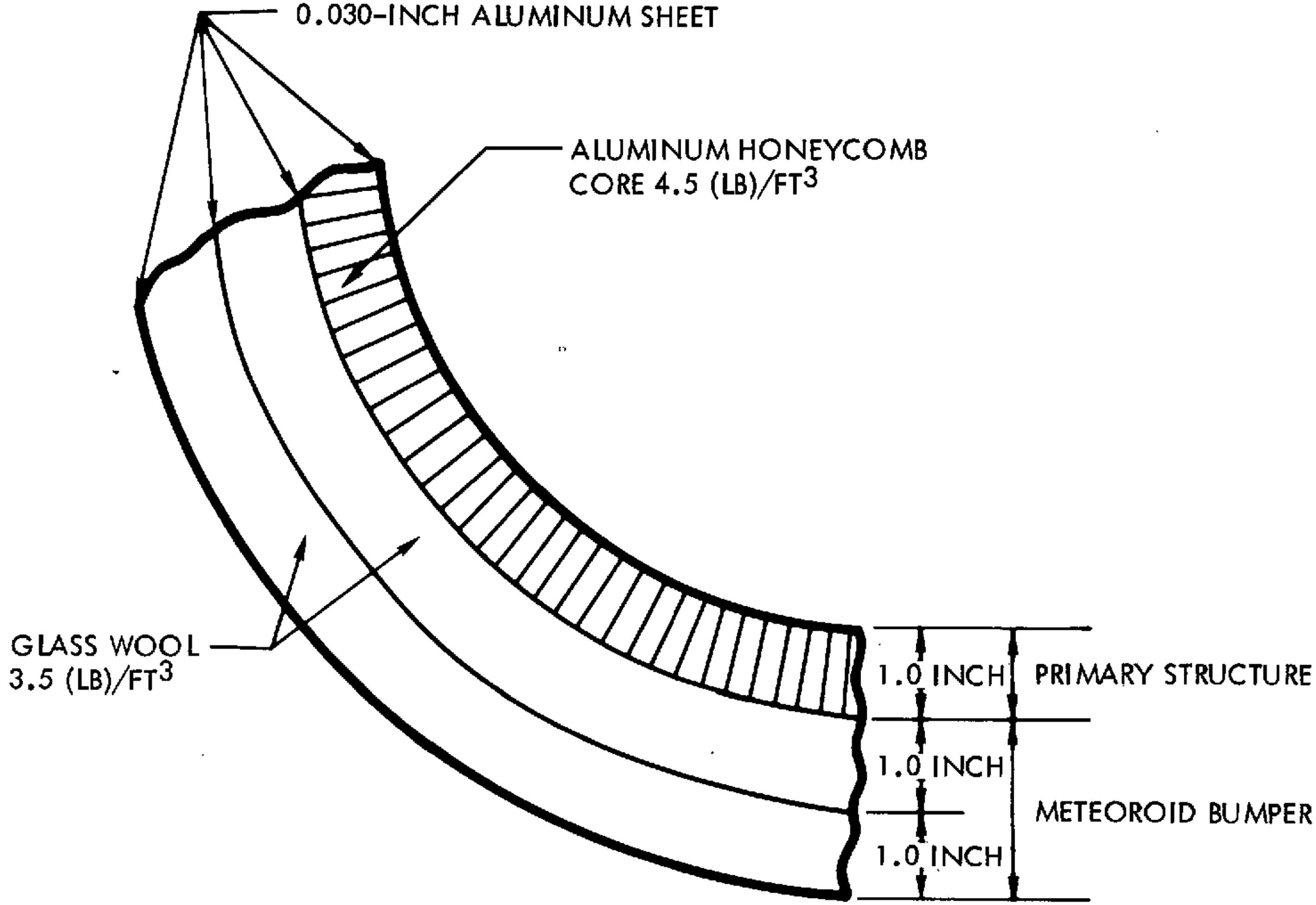


Figure 12. Rigid Module Design

stress level allow for a redistribution of loads around a puncture so that the propagation of a crack from a hole will be less likely to occur. This bumper design was based on the evaluation of data from References 3 and 4. It was estimated that 240 penetrations a year of the outer structural wall would occur; and of these, 145 meteoroids would penetrate the inner wall.

Additional information on the meteoroid flux and penetration data has brought about a review and re-evaluation of the meteoroid bumper and module wall design. The re-evaluation resulted in a reduction of the wall structural weight by 23 percent as well as a reduction in the probability of penetration. The equations presented in Appendix A can be used to determine the total skin thickness required for multiple-sheet structures. They were derived by combining the best estimate of meteoroid flux with penetration equations and assigning an efficiency factor, based on emperical data, for different multiwall structures. Figure 13 is a plot of the number of penetrations per year on a vehicle with a surface area of 20, 000 square feet versus weight of single-sheet structure and a two-, three-, and four-sheet structure. It can be seen that there is little or no weight benefit by going to a four-sheet structure, in fact if the joining and attachment weights were considered, the four-sheet structure would be the heavier. This curve was based on Whipples 1957 meteoroid flux. The curve of the number of penetrations per year versus weight for the three-sheet structure, Figure 13, shows that a large increase in weight above 40, 000 pounds gives only a small reduction in the number of penetrations.

For example, going from a structural wall weight of 30, 000 pounds to 40, 000 pounds reduces the number of penetrations from 40 to 13 per year. Whereas, going from 40, 000 pounds to 50, 000 pounds reduces the number of penetrations from 13 to 5 per year. This plot then shows the best design point, considering both weights and number of penetrations, to be approximately that of 40, 000 pounds and 13 penetrations per year.

The total wall thickness was determined by the following equation:

$$S_m = \frac{1.97}{F^{1/3}\, E^{2/3}}$$

where S_m = total wall thickness in inches

F = number of penetrations per year

E = efficiency of multiwall structure

This equation was derived by combining equations 3, 11, and 14 of Appendix A. The efficiency factors are listed in Table 2 of Appendix A.

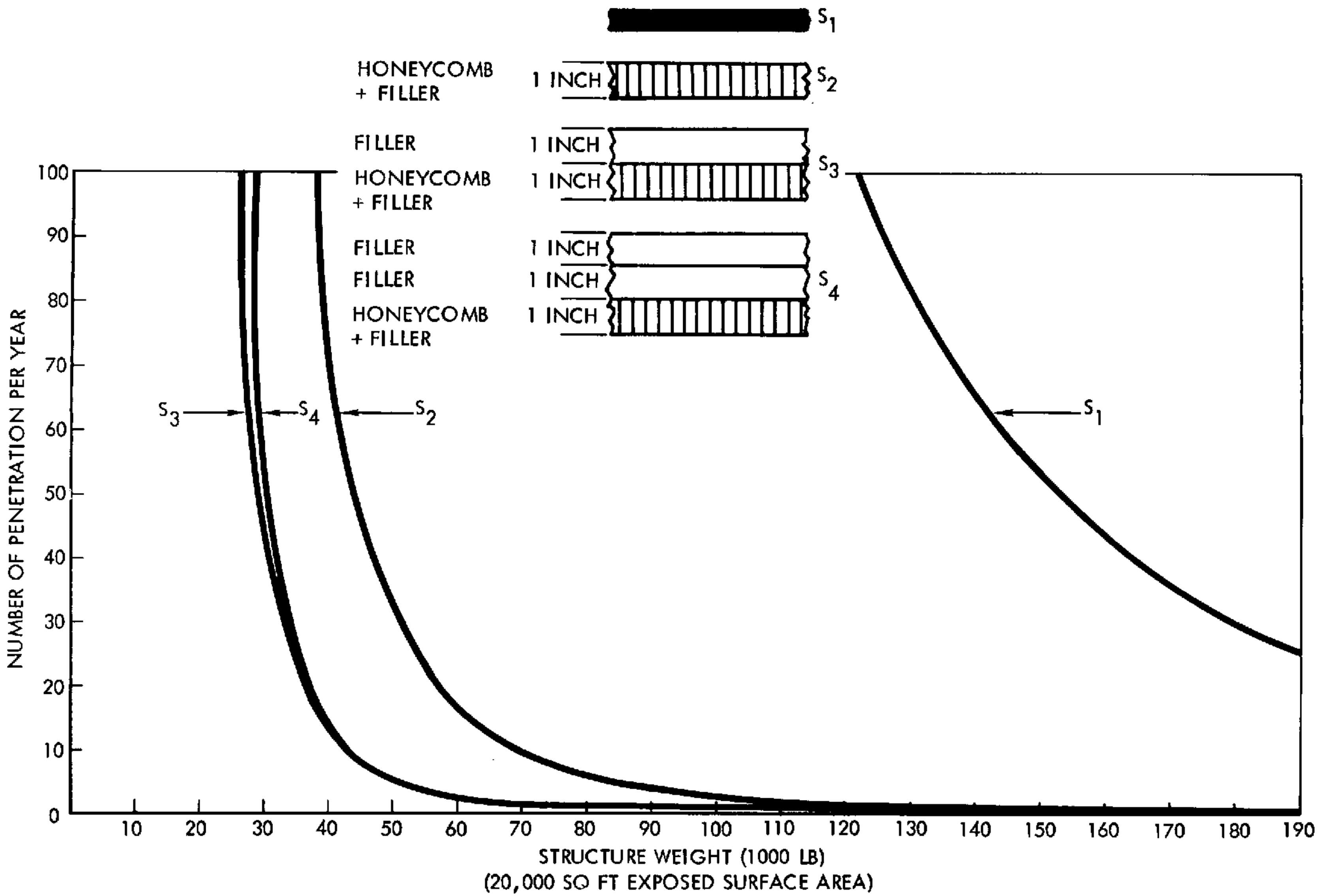


Figure 13. Structure Weight Versus Meteoroid Penetrations (Whipple 1957 Meteoroid Flux)

This equation was also used to determine the skin thickness weight used in constructing Figure 13. Substituting in the equation, the total aluminum thickness is

$$S_m = \frac{1.97}{(13)^{1/3}\,(25.9)^{2/3}} = \frac{1.97}{2.35 \times 8.75} = 0.096 \text{ inches}$$

This total thickness is then distributed between the bumper sheet and primary structure sheet by choosing a standard production gage of 0.020 inches for the bumper and finding that it falls within the 15- to 50-percent weight distribution requirement shown in Appendix A. The remaining 0.076 is then divided into equal thickness sheets for primary structure which makes them 0.038 inches each. A 1-inch spacing is used between sheets. The first space, from inside out, contains aluminum honeycomb material and when bonded to the face sheets constitutes the primary load-carrying structure. Penetration tests have shown that honeycomb between sheets tends to channel the striking particles and reduces the efficiency of protection gained by the spacing. Polyurethane filler has been found to be an efficient protection device, so the honeycomb is filled with polyurethane and assumed to balance the channeling effect giving the structure the same efficiency as two sheets with a void between them. The remaining space is filled with polyurethane and, in conjunction with the 0.020 inch aluminum sheet, performs the function of a meteoroid bumper. This given the module wall design shown in Figure 14.

The stress analysis then for the final configuration is performed using this wall design as the basic structure. Additional longerons (six longerons are required to join the wall structure panels due to production size limitations) were required to withstand high local bending moments in the boost configuration. This becomes an area for further study in that a change in support points of the modules in the boost configuration will change the type and magnitude of the loading and require a weight trade-off study to determine a minimum weight arrangement.

Data from additional meteoroid experiments were received and evaluated after the structure for the final configuration was established. This data has brought about a change in the meteoroid flux equation and can result in a further weight reduction for meteoroid protection. Figure 15 shows the number of penetrations per year versus weight, incorporating the change in meteoroid flux. The equation used for this comparison is

$$S_m = \frac{0.751}{(F)^{0.234}\,(E)^{0.667}}$$

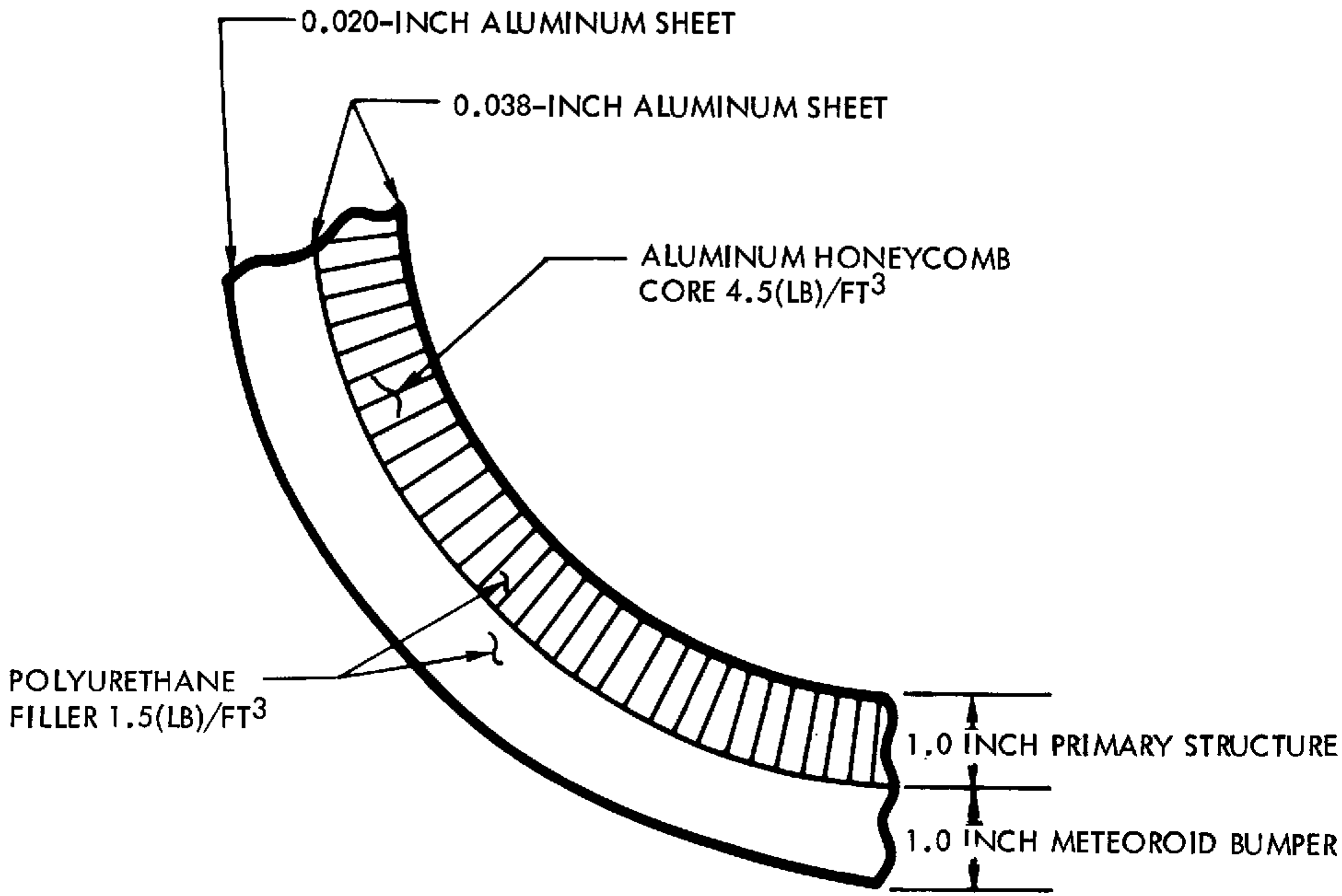


Figure 14. Meteoroid Bumper and Module Wall Desig-

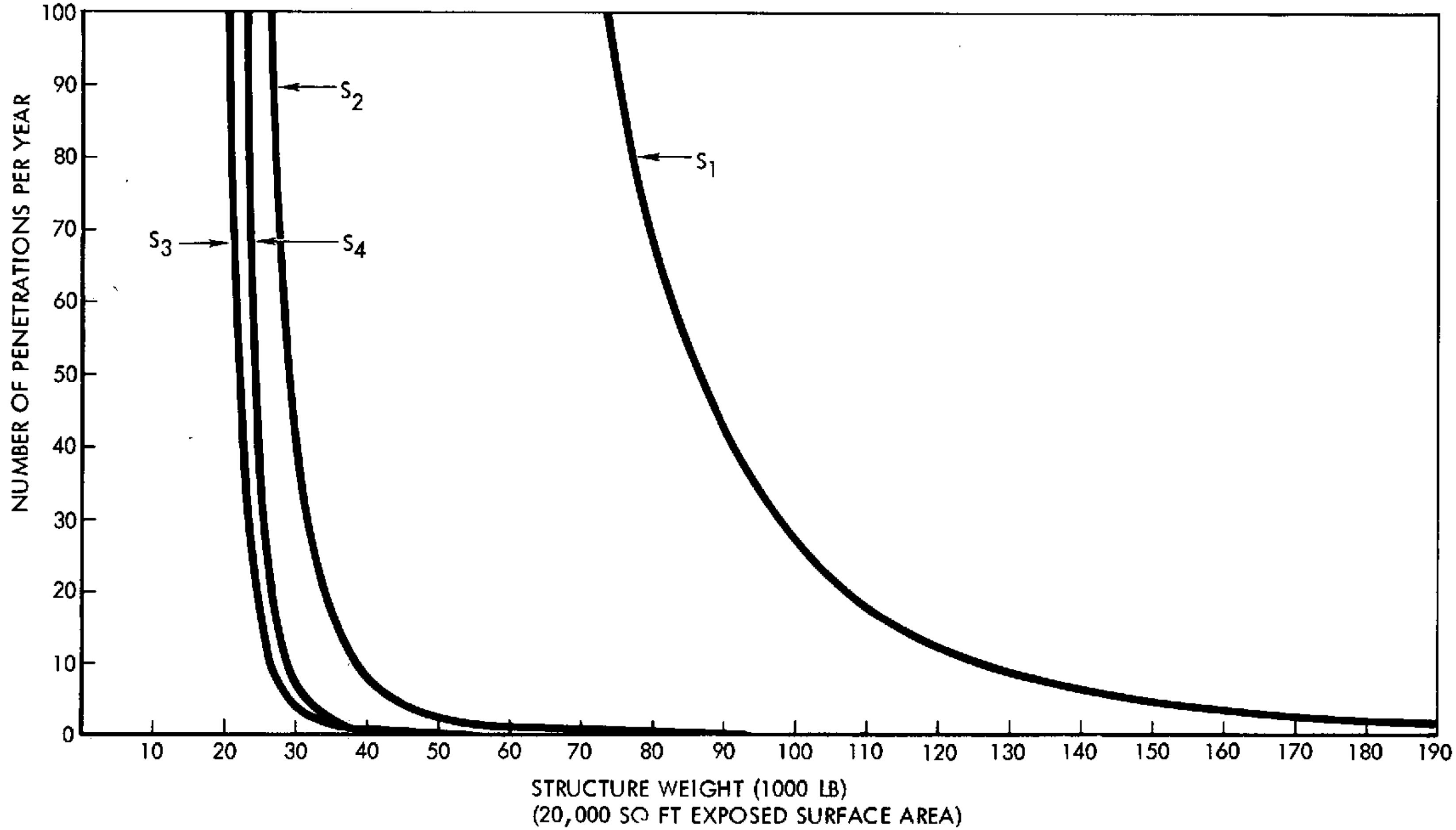


Figure 15. Structure Weight Versus Meteoroid Penetration

This equation is derived by combining equations 14 and 19 of Appendix A. The wall design then becomes an area for further study since a reduction in wall skin thickness for meteoroid protection will raise the stress levels and possibly show the strength requirement to be more critical than the meteoroid protection requirement.

Types of Construction

The honeycomb-sandwich type of construction used in the rigid module was established on the basis of meteoroid protection requirements and is discussed in the previous section. A study of four types of construction was performed on the basis of strength requirements in order to arrive at the type of construction that gives minimum structural weight when the strength requirements become the critical design criteria. The four types of construction considered in the analysis of the space laboratory are sandwich, longerons, and frames; sandwich and frame; and two types of semimonocoque construction. These are skin and frame and skin, stringer, and frame. Figure 16 shows the types of construction.

This analysis was performed on a comparative basis by using 2014-T6 material operating at a temperature of 300 F. This is less than the 350 F temperature encountered during trajectory due to aerodynamic heating; however, the actual structure will be operating at an even lower temperature since the meteoroid bumper also provides thermal insulation. In all cases, frames are provided to prevent buckling of the walls under high compression loading. The frame stiffness used to prevent the general instability mode of failure is determined from the empirical relationship:

$$EI = \frac{MD^2}{16,000\ L}$$

Where

E = modulus of elasticity of frame material

I = moment of inertia of frame

M = equivalent bending moment

D = diameter

L = frame spacing

This expression is derived from tests on stiffened shells in pure bending (Reference 6); so the combined axial and bending load is converted to a total equivalent bending moment.

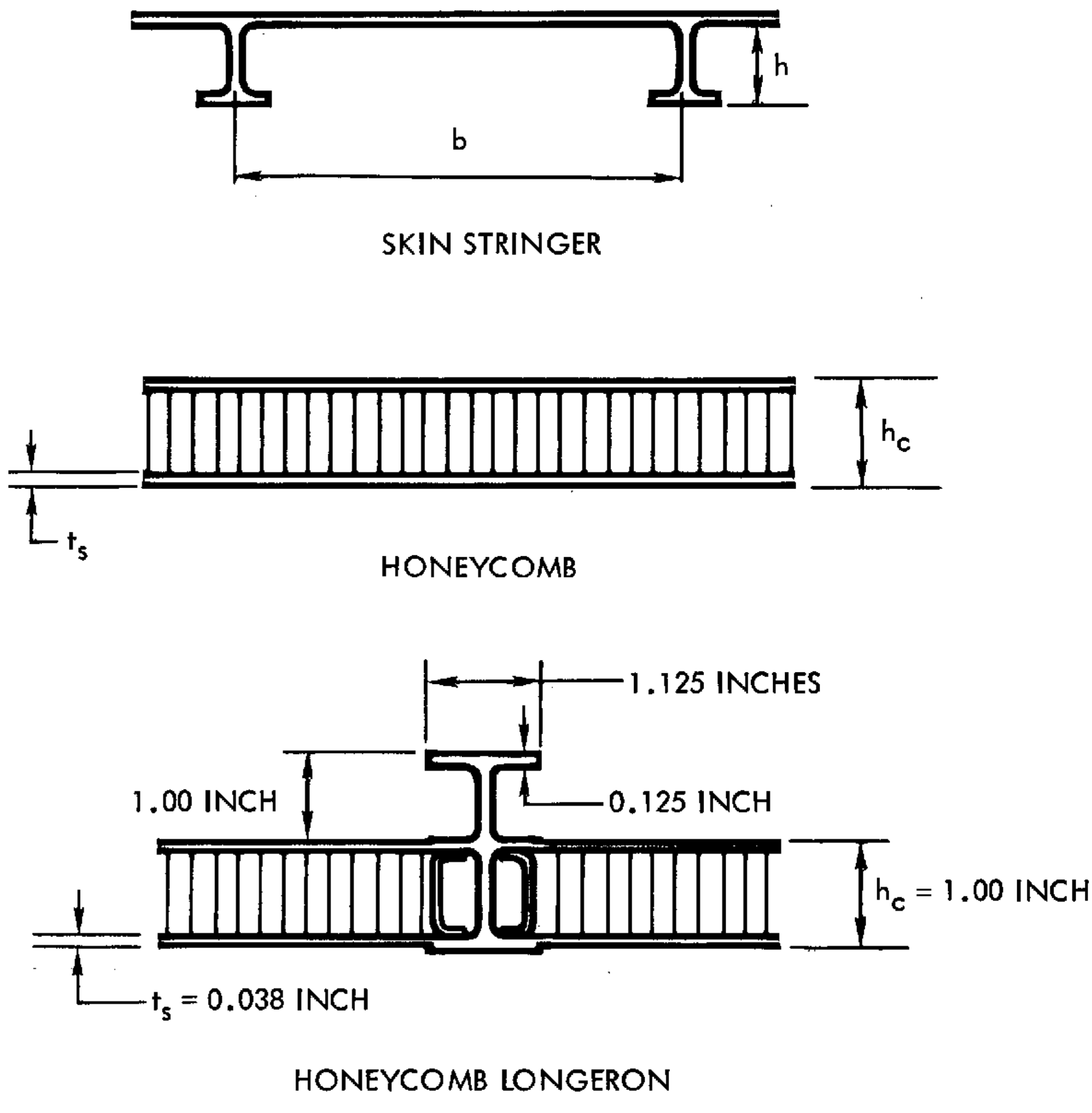


Figure 16. Types of Construction

In the skin-frame type construction, all the primary loading is carried in the skin. The critical buckling allowable stresses were obtained from the following relationship from Reference 5:

$$\frac{\sigma_a}{\eta} = K_c \frac{\pi^2 E}{12(1 - \mu^2)} \left(\frac{t}{L}\right)^2$$

where

σ_a = allowable buckling stress

η = plasticity correction

K_c = buckling coefficient

E = modulus of elasticity of skin material

μ = poisson ratio

t = skin thickness

L = frame spacing

Due to the low allowable buckling stress and large surface area of the space station, the relative weight of skin-frame construction, as shown in Figure 17, is quite high when compared to the other types of construction.

The skin and stringer type of construction provides a more efficient design by the selective arrangement of the structural material in the stiffener and in the skin. The stringer sectional area, proportions, and spacing are used with numerous skin gages to determine the optimum composites strength to weight comparison.

The 2014-T6 aluminum honeycomb being considered in the analysis consists of the two facing sheets and core intimately fixed in relation to each other by a bonding agent. This allows the facing sheets to sustain primary loading in its plane while the core provides lateral support.

The column allowable stress was modified to account for shear stiffness of the core by the following emperical relationship from Reference 7:

$$P_a = \frac{P_1 G_c h_c}{P_1 + G_c h_c}$$

where

P_1 = allowable column stress

P_a = modified column stress

G_c is core shear modules

h_c is core depth

Optimum sandwich configuration was determined by obtaining the proper proportions of core depth and shear modulus for the corresponding face sheet gages as shown in Figure 18.

The fourth type of construction consists of the honeycomb panel reinforced with stringers. The composite section shown in Figure 16 has been analyzed considering the honeycomb panel as an equivalent homogenous skin stiffness with stringers with frames. This type of construction has the best strength-to-weight ratio for high unit loadings and is used in the modules when the loading dictates.

Analysis of Number of Modules

An evaluation of four arrangements of the vehicle structural components in the boost configuration was made, in terms of relative weight, for design support in selecting the optimum number of modules and arrangement of components. These four arrangements are shown in Figure 19. The zero-g laboratory, hub, and dock are located adjacent to the booster on arrangements 1 and 2. They differ in that the laboratory is connected to the upper fairing in arrangement 1 and not connected in arrangement 2. The zero-g laboratory, hub, and dock are located at the top of the modules on arrangements 3 and 4. These two arrangements differ in that the laboratory is connected to the lower fairing in arrangement 3 and not connected in 4. Varying the number of modules from four to six, six to eight, and eight to ten determined the basic parameters of module length and overall length of the vehicle.

Due to the numerous parameters involved in the study, several considerations and assumptions were made for the expediency and simplification of the analysis. These are as follows:

1. Total launch weight of each configuration is approximately 150, 000 pounds.

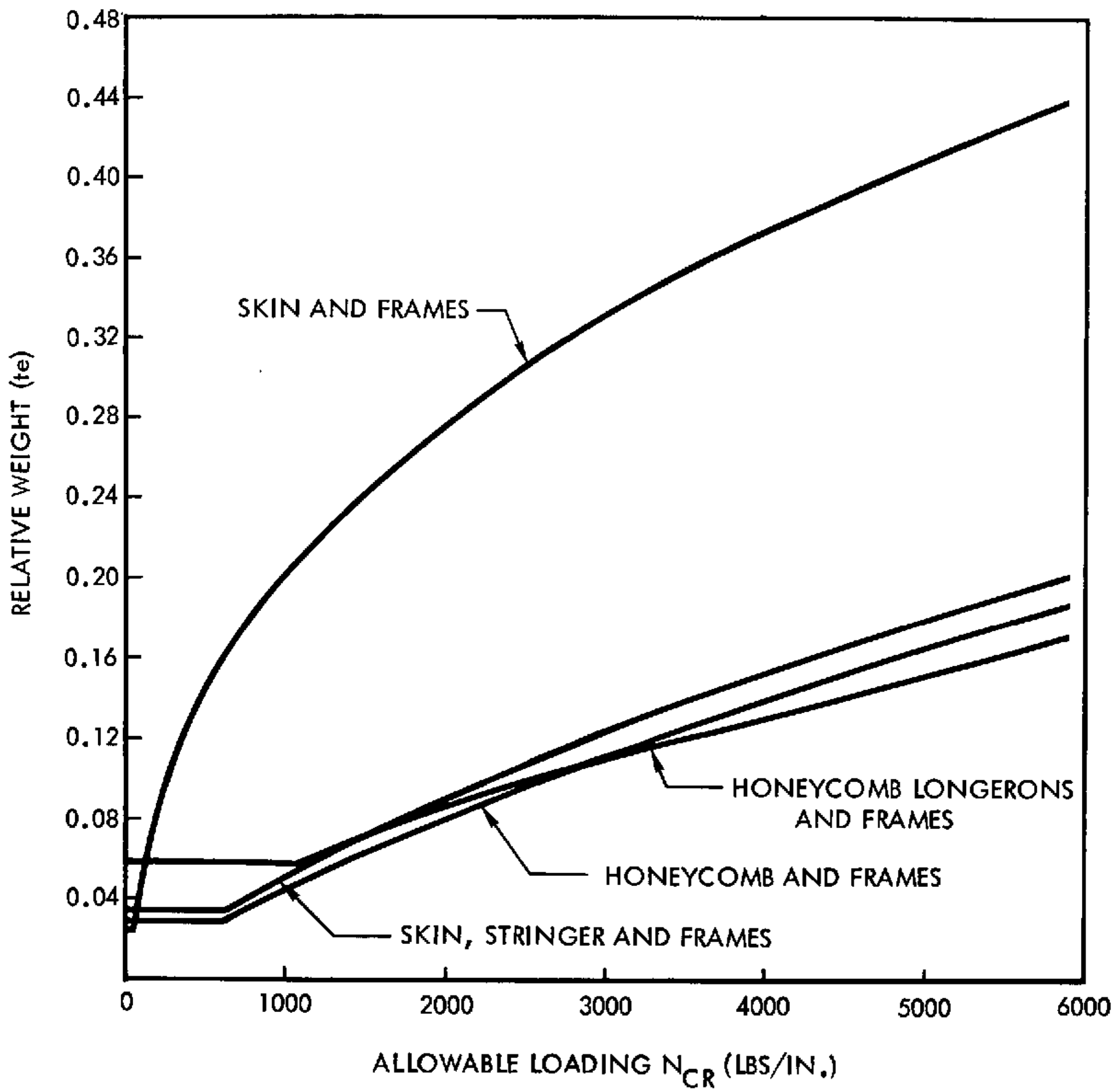
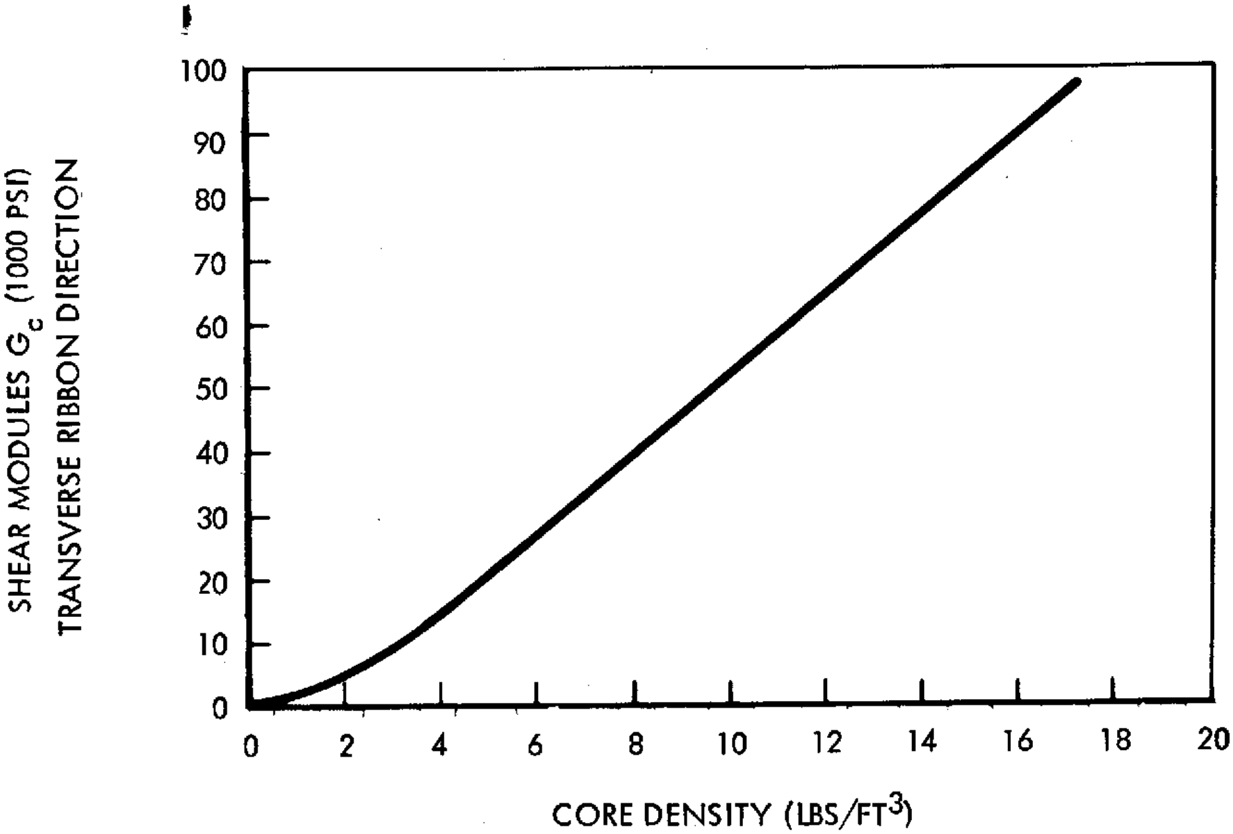


Figure 17. Types of Construction - Relative Weight



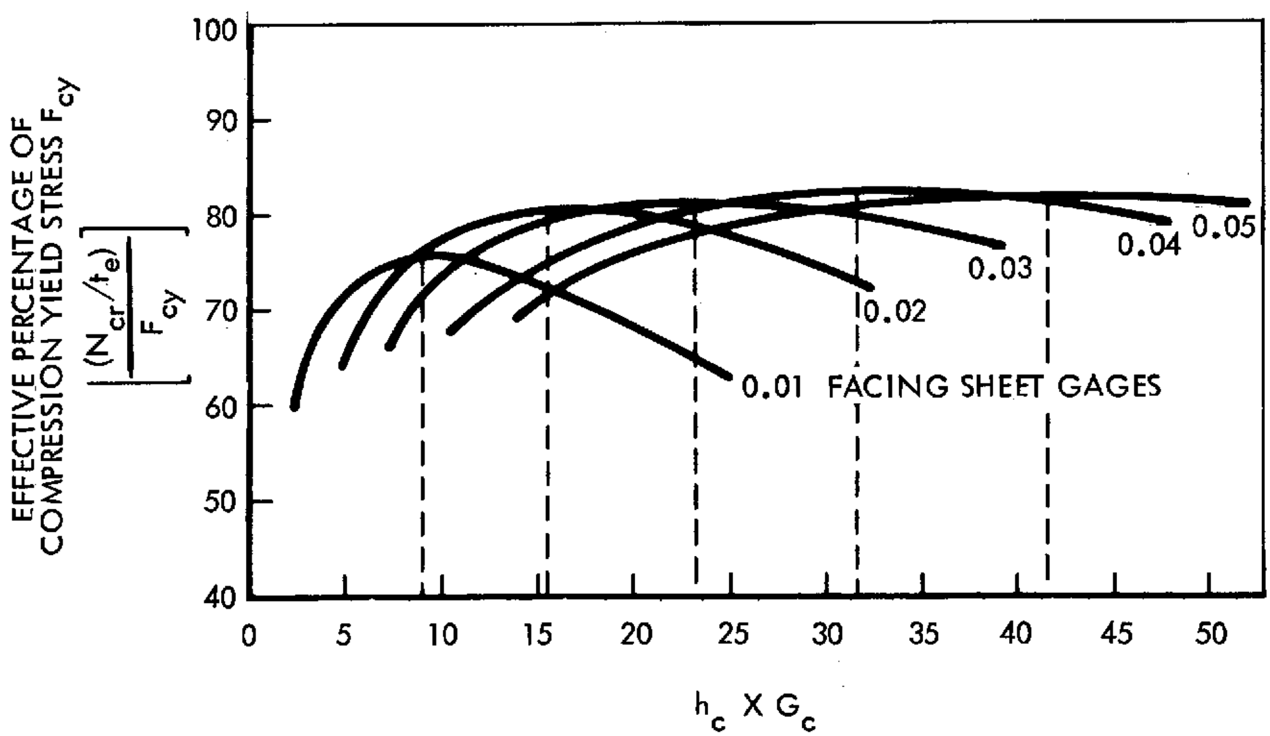


Figure 18. Shear Modulus Versus Core Density

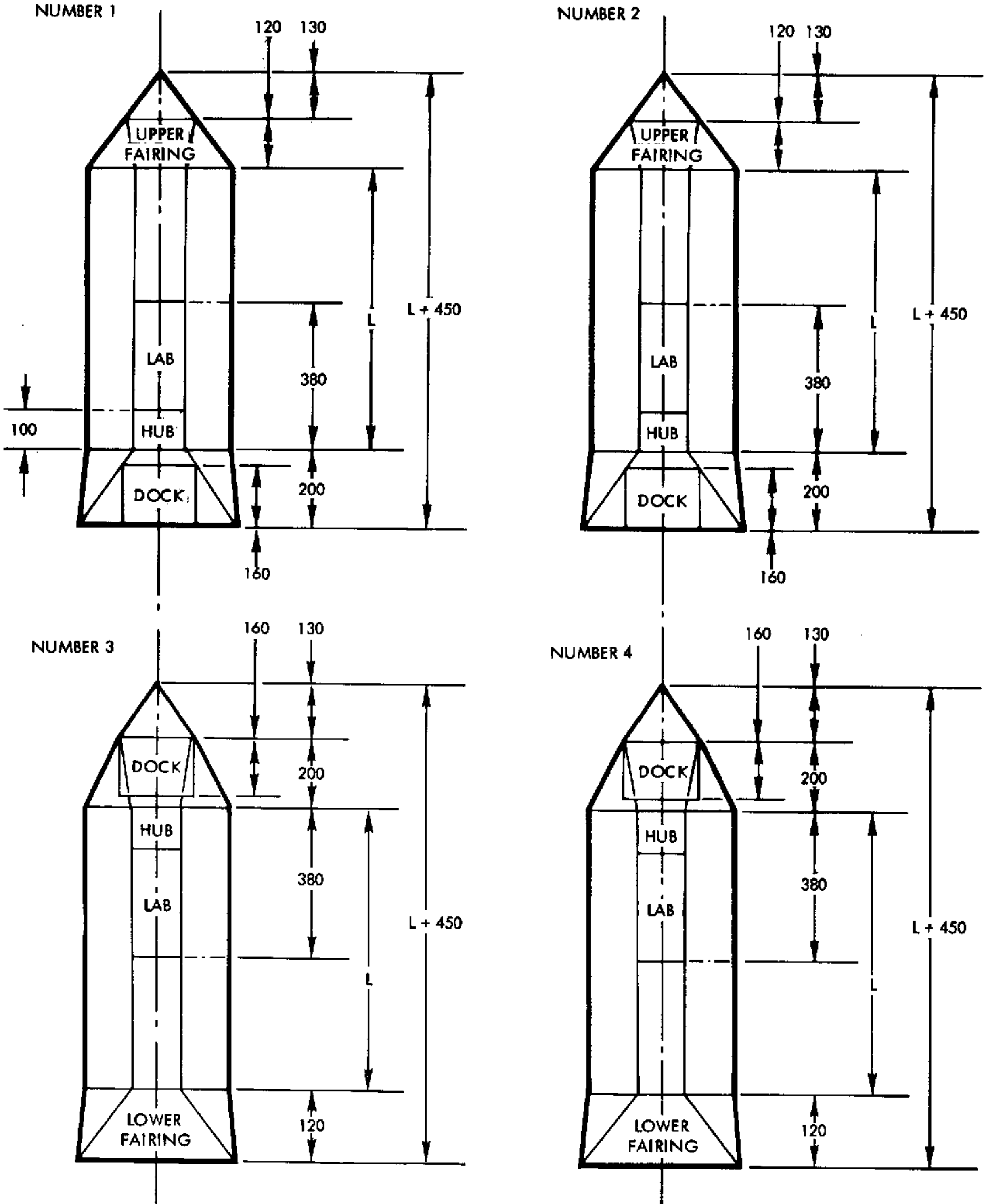


Figure 19. Configuration Analysis

2. The dock, zero-g laboratory, and hub consist of the same shape, volume, and structural weight for meteoroid protection; however, they are arranged in a different manner as noted above. The basic geometry used in this study is as follows:

   Dock - 170 inch diameter by 160 inches long

   Connecting hub - 120 inch diameter by 100 inches long

   Laboratory - 120 inch diameter by 280 inches long

   Modules - 120 inch diameter

   Packaged overall lengths and corresponding number of modules are the same for each configuration

   Upper fairing diameter = 170 inches (attachment to Apollo)

   Lower fairing diameter = 396 inches (attachment to S-II stage)

3. The modules are rigidly fixed to the upper and lower fairings.

4. The 3-inch, four-sheet aluminum module wall designed for meteoroid protection was used as the basic structure (Figure 12).

5. Each module has two end bulkheads with airlocks.

The upper and lower fairings of the central hub form a cluster attachment for the modules. The weight of these fairings is included in the hub weight in summary Table 2. Arrangements 1 and 2, with the dock adjacent to the booster, required the addition of structural material to the dock above meteoroid protection requirements in order to carry the loads. The upper fairing was relatively light. Arrangements 3 and 4, with the dock on top of the modules, required no additional dock structure but a relatively heavy lower fairing. A combination of these factors was required to obtain the best weight.

The design stresses for boost trajector loads did not exceed the requirements for meteoroid protection and internal pressure, for arrangements of six modules or more. The basic difference in torus weight for six or more modules is in the number of end bulkheads with airlocks. This is due to an increase in bending moment resulting from the increased boost length between the four and six module arrangement.

The results of this evaluation are shown in Table 2 as a relative weight summary showing the corresponding weight distribution between the torus and

Table 2. Relative Weight Summary

Configuration Number	Item	Number of Modules 4	6	8	10
1	Hub	10,915	10,928	10,585	11,402
	Torus	32,000	29,400	29,850	30,400
	Total	42,915	40,328	40,435	41,802
2	Hub	10,535	9,908	10,535	11,352
	Torus	33,000	29,400	29,850	30,400
	Total	43,535	39,308	40,385	41,752
3	Hub	11,689	10,784	10,525	11,362
	Torus	32,300	29,400	29,850	30,400
	Total	43,989	40,184	40,375	41,762
4	Hub	10,119	9,984	10,575	11,412
	Torus	34,650	29,400	29,850	30,400
	Total	44,769	39,384	40,425	41,812

central hub and the total comparative weight of each arrangement. The totals are plotted in Figure 20 as relative structural weight versus number of modules, and shows the structural weight to be minimum for all arrangements of six modules. Although arrangement 2 shows the least weight, the greatest weight for six modules (arrangement 1) is only 3 percent more. Thus, the structural arrangement has little effect on the overall structural efficiency for a vehicle of equal number of modules.

Since this analysis was completed, the structural requirements for meteoroid penetration has been reduced. This change will have a significant effect on the results of this study. It is noted that a sizable weight benefit is realized by considering the modules rigidly attached to the upper and lower fairings. This benefit, however, may penalize deployment reliability. Therefore, deployment reliability and functional operation may play an important part in determining the optimum configuration.

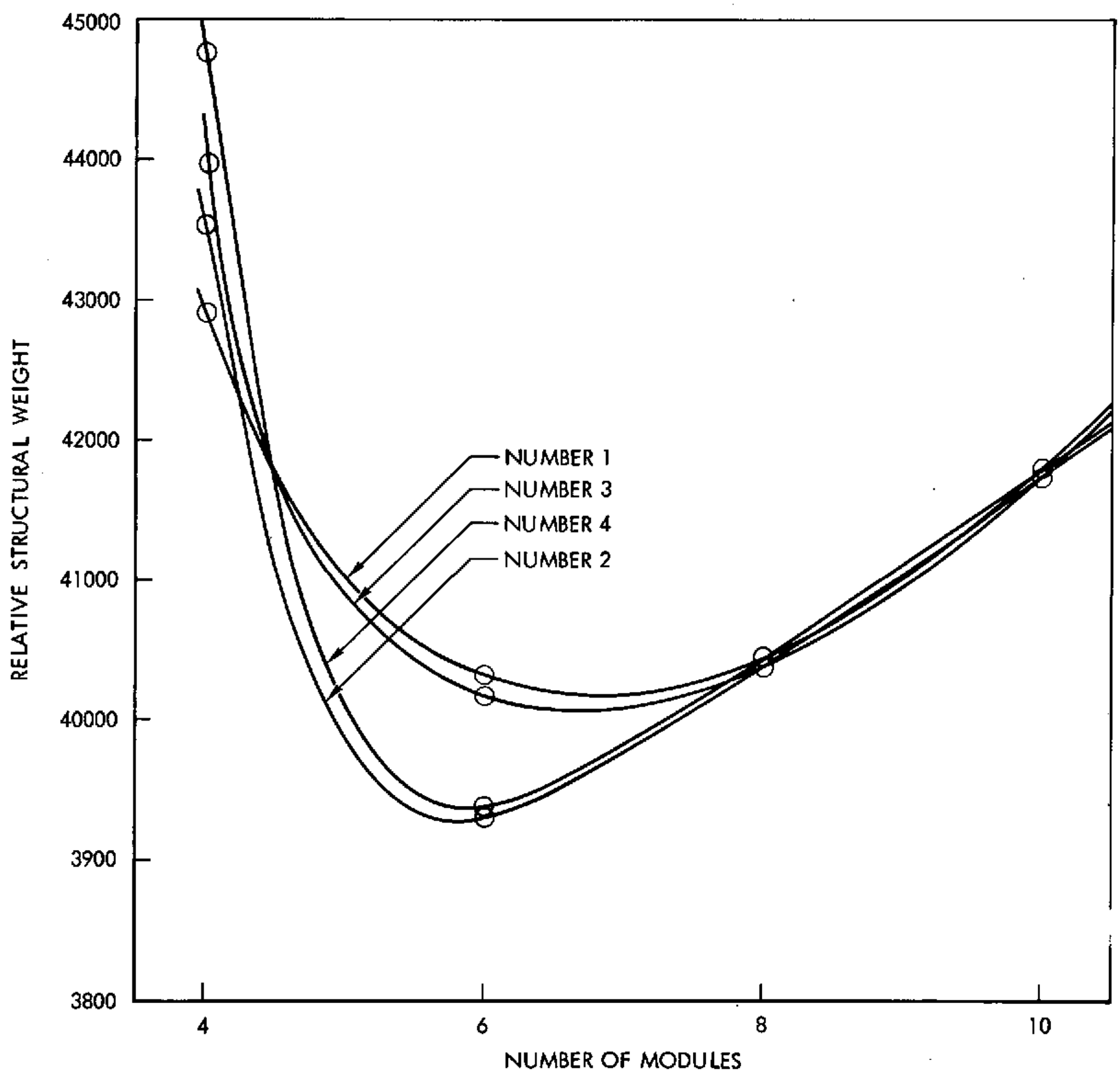


Figure 20. Configuration Analysis - Relative Weight

Thermal Stress Analysis

A brief thermal stress investigation was made in order to determine if the estimated heating rates and temperature gradients would result in a severe weight penalty. The heating rates and thermal gradients were estimated for the boost condition and were taken from the environmental system analysis for orbital flight (Table 3). The module wall used in this analysis is the 3-inch, four-sheet aluminum wall shown in Figure 12.

During the boost cycle, the outer surface fibers experience a moderately high heating rate of approximately 1 F per second imposed upon the central hub fairings. A much lower heating rate is anticipated on the clustered modules because of a lower external skin temperature. The equivalent body force system on the central hub fairing shown in Figure 21 will then be determined for the radial temperature gradient. For the short holding time estimated during boost a radial temperature gradient, T = 280 F will exist between the inner and outer central hub fairing wall. The outer surface compressive stress necessary for the entire suppression of thermal expansion is obtained from Reference 8:

$$\sigma = \frac{E\alpha\Delta T}{2(1 - \mu)\log \frac{b}{a}} \left[1 = \frac{2a^2}{b^2 - a^2} \left(\log \frac{b}{a}\right) \right]$$

where

σ = induced thermal stress

E = modulus of elasticity of material used

α = coefficient of thermal expansion

ΔT = radial temperature difference

μ = Poisson's ratio

b = radius to outer surface

a = radius to inner surface

The outer surface compressive stress and the inner surface tensile stress is approximately 25, 000 psi.

This stress level when combined with other stresses imposed during the boost condition is not expected to impose an appreciable weight penalty on the fairing. The detailed effects of thermal stresses during the boost

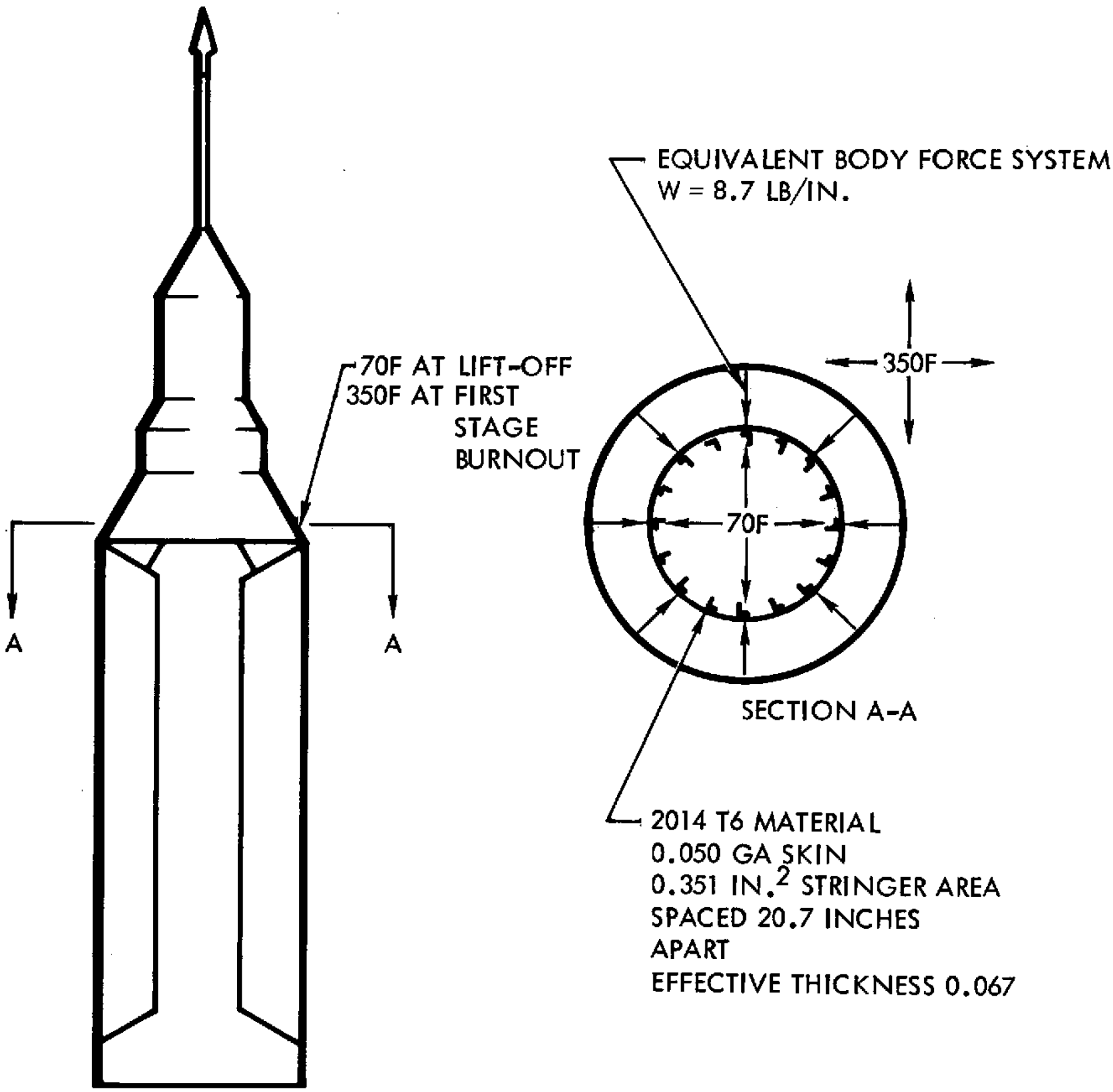


Figure 21. Boost Configuration Radial Gradient at Central Hub and Equivalent Body Force System

Table 3. Temperature Distribution

Time Interval (Min.)	Module Elements*											
	1	2	3	4	5	6	7	8	9	10	11	12
	Absorptivity (α)											
	0.3	0.3	0.5	0.4	0.3	0.3	0.3	0.3	0.4	0.5	0.3	0.3
	Emissivity (ϵ)											
	0.9	0.9	0.5	0.1	0.1	0.1	0.1	0.1	0.1	0.5	0.9	0.9
	Temperature (F)											
0	73	42	64	28	38	60	56	28	11	33	25	67
10	75	40	68	34	51	78	77	47	29	52	35	74
20	81	41	66	30	49	78	80	53	38	71	48	82
29.2	87	45	63	23	42	69	71	48	33	75	61	91
40	27	-3	24	16	33	58	60	39	27	37	16	32
50	4	-23	-2	10	25	49	51	31	20	12	-7	9
60	-6	-32	-16	5	18	39	42	23	14	-11	-28	-10
65	-9	-34	-20	3	16	34	38	20	10	-22	-42	-21
70	17	-1	-3	1	13	30	34	16	7	-12	-20	+12
80	55	28	32	5	15	32	30	12	2	6	7	49
90	69	41	56	19	29	48	43	20	6	21	19	64
95+	72	42	64	28	40	61	58	30	13	32	24	68

*See Figure

cycle requires further investigation when trajectory data and surface temperatures of the packaged configuration are more accurately defined.

During the orbital flight cycle, the outer surface fibers experience a relatively low heating rate, in the order of 0. 040 F per second. The induced thermal stresses developed at the outer surface fibers due to the expansion constraint would not be in excess of 10, 000 psi at 70 F. The quenching or cooling rate is more significant, approximately 0.065 F per second, resulting in the order of 13,000 psi thermal stress at low temperatures approximately -42 F. The thermal-stress fatigue due to cycles of temperature fluctuations do not appear to be critical; however, these effects should be investigated further.

An analysis of thermal stress was performed on the modules using the maximum thermal gradient experienced during the orbital time history to obtain an equivalent body force system for determining the actual induced stresses. Figure 22 shows the equivalent body force system for this temperature gradient. The outer surface tension stress and inner surface compressive stress for this condition is approximately 1000 psi. The computations are shown in SID 62-615, Preliminary Stress Analysis.

FINAL CONFIGURATION ANALYSIS

The final configuration of the space station consists of four major structural components, central hub, modules, spokes, and thrust structure as shown in Figure 23. The analysis of this configuration consists of determining the boost and orbital flight loads; establishing geometries and allowable stresses for frame and shell materials; and determining structural member sizes in the hub, spoke, module, and thrust structure. Module joint discontinuity stresses were also determined. This discussion presents a brief summary of the methods used and the results of this analysis. The computations are presented in SID 62-615.

Loads

The basic load analysis consists of applying the pressures and load factors, previously determined in the design criteria, to the structure and the distribution and transmission of these forces through the structural components. Because of the environmental conditions involved, the load analysis is separated into two categories, ground and boost trajectory conditions and orbital flight conditions.

Ground and Boost Trajectory Body Loads

The space station is packaged for the ground prelaunch and boost trajectory loading conditions as shown in the basic geometry Figure 23.

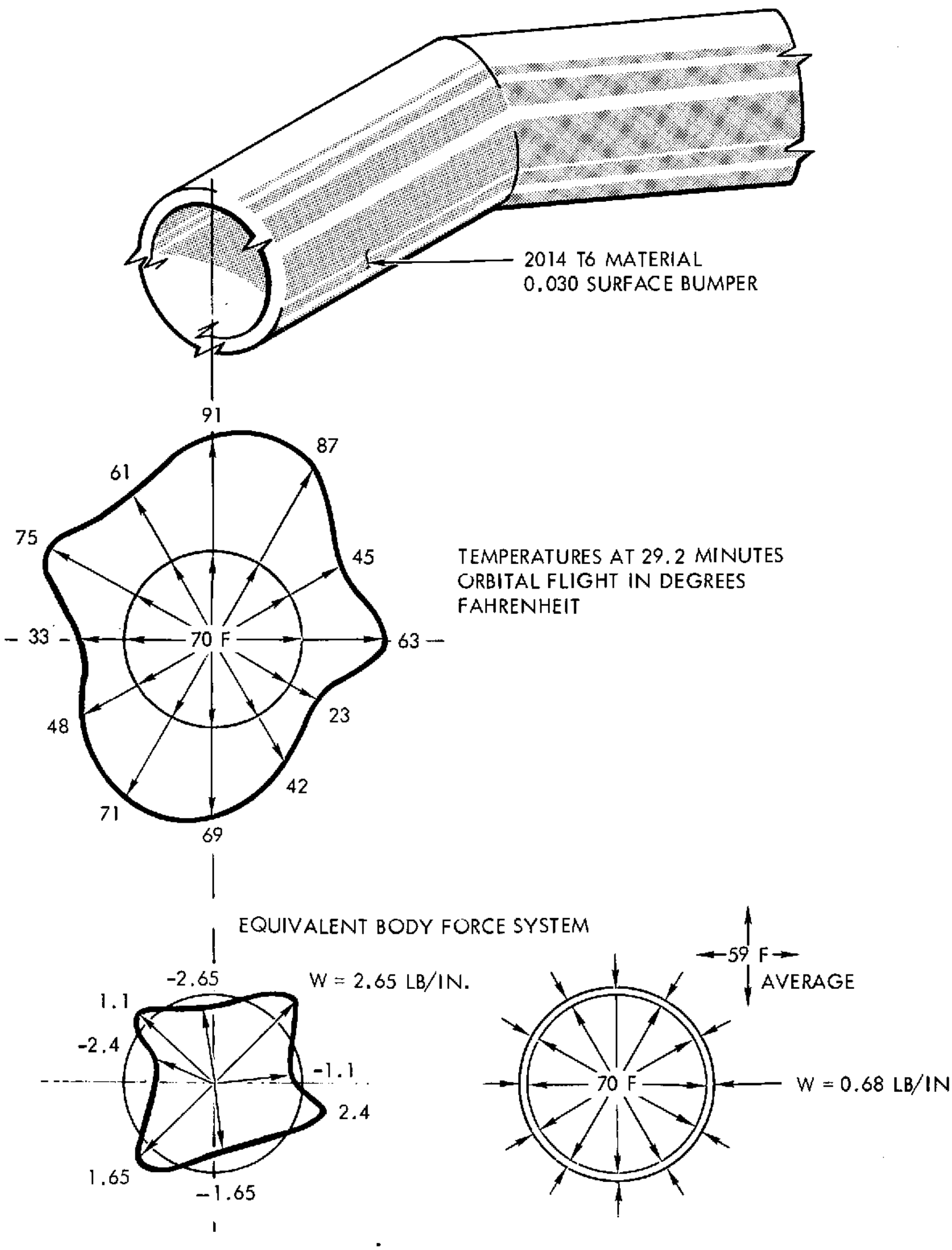


Figure 22. Orbital Flight Conditions

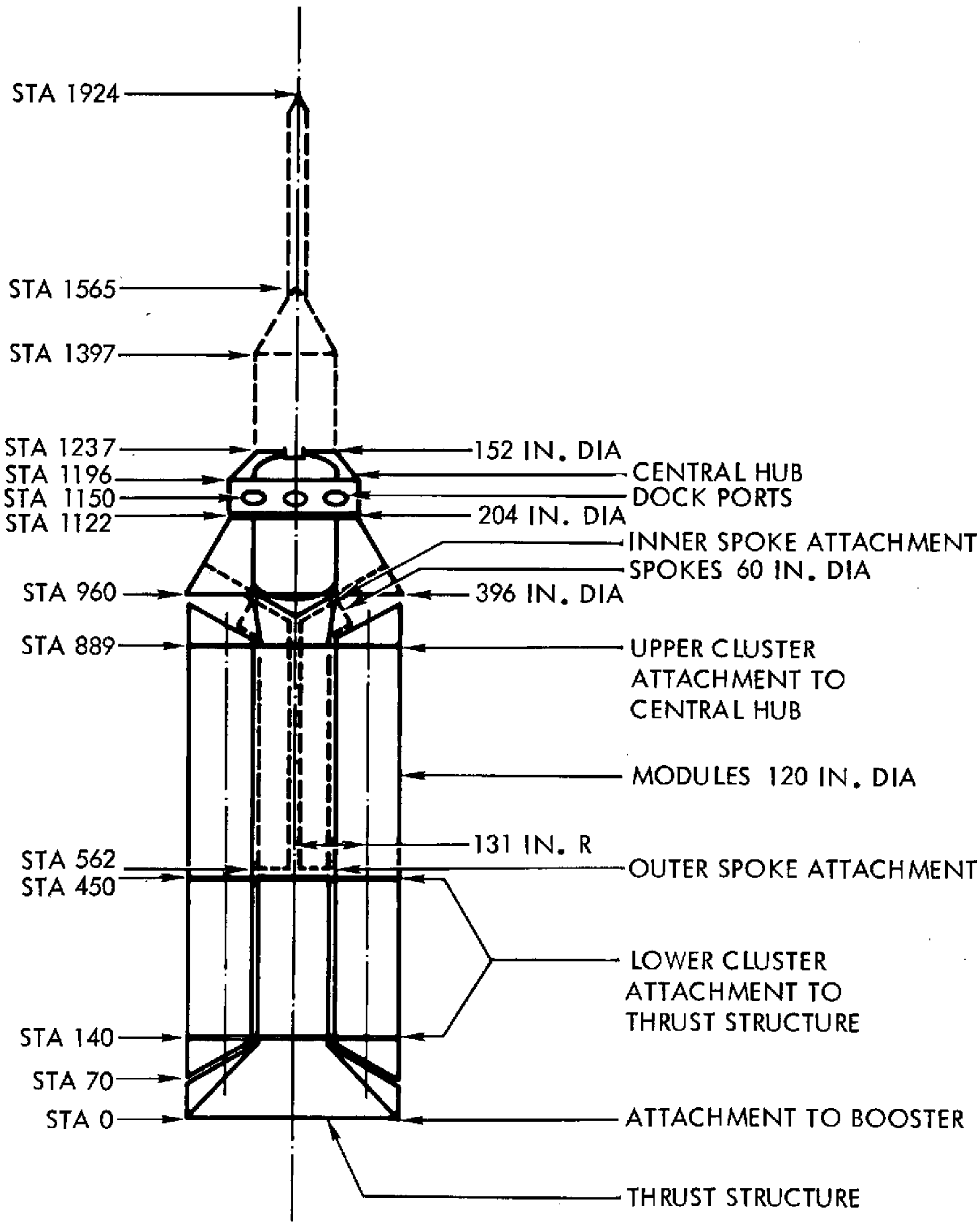


Figure 23. Geometry in Ground Prelaunch and Boost Trajectory Conditions

Attachment fittings are located at the lop of the module to transfer the load from the central hub assembly to the modules. To transfer the axial and bending loads from the modules to the thrust structure there are fittings located at stations 450 and 140. For simplicity, unit conditions of inertia and airload (1 g and 1 psi respectively) are computed and distributed to the structural system as shown in Figure 24 for air-load and Figure 25 for inertia load. Thus, for each specific loading condition, load factors and actual airloads are superimposed as multiples of the unit conditions. It may be noted that the attachment fittings are assumed to be virtually pin jointed to provide a more reliable deployment function. Restraining moments in the fittings may be further investigated in order to obtain a more precise load analysis. This support arrangement required additional longeron material in the module walls to withstand high local bending moments. This becomes an area for further study in that a change in support points of the modules at the thrust structure will change the type and magnitude of the loading and require a weight trade-off study to determine a minimum weight arrangement. A summary table of unit loads and actual loads are shown in Tables 4 and 5 and provides the design envelope of the ground and trajectory loading conditions with the corresponding environmental conditions.

The boost inertia loading conditions were computed using the estimated structural weight distribution shown in Table 6. The launch weight on the group weight statement is now 13 percent higher; therefore, the inertia loads will increase by this amount. At the beginning of the analysis, the packaged payload was 1924 inches long from the booster-payload interface. The Apollo service module has since been shortened by 104 inches, reducing the body bending moments approximately 10 percent. Since the effect of body bending on the structure is more critical than the axial loading, the body loads are slightly conservative. The actual boost loading conditions may be refined in future investigations.

Orbital Flight Body Loads

The space station is deployed, pressurized, and rotated to provide artificial gravity. The loads encountered during deployment are considered to have only local effects, and for this reason are not included in the study. The loads encountered during station operation will be an internal pressure of 10 psi and a radial force up to 1 g on the rim.

Control jets are located along the rim of the station to produce spin-up and pitching control forces, Figure 26. These forces will induce small concentrated loads on the modules and are not considered in this study. Body loads are computed for unit spin-up and pitching accelerations. The loads produced by the actual accelerations were found to be negligible. The thermal loads, worked out for the 3-inch wall construction, were not repeated for the final configuration analysis because they produced small stresses.

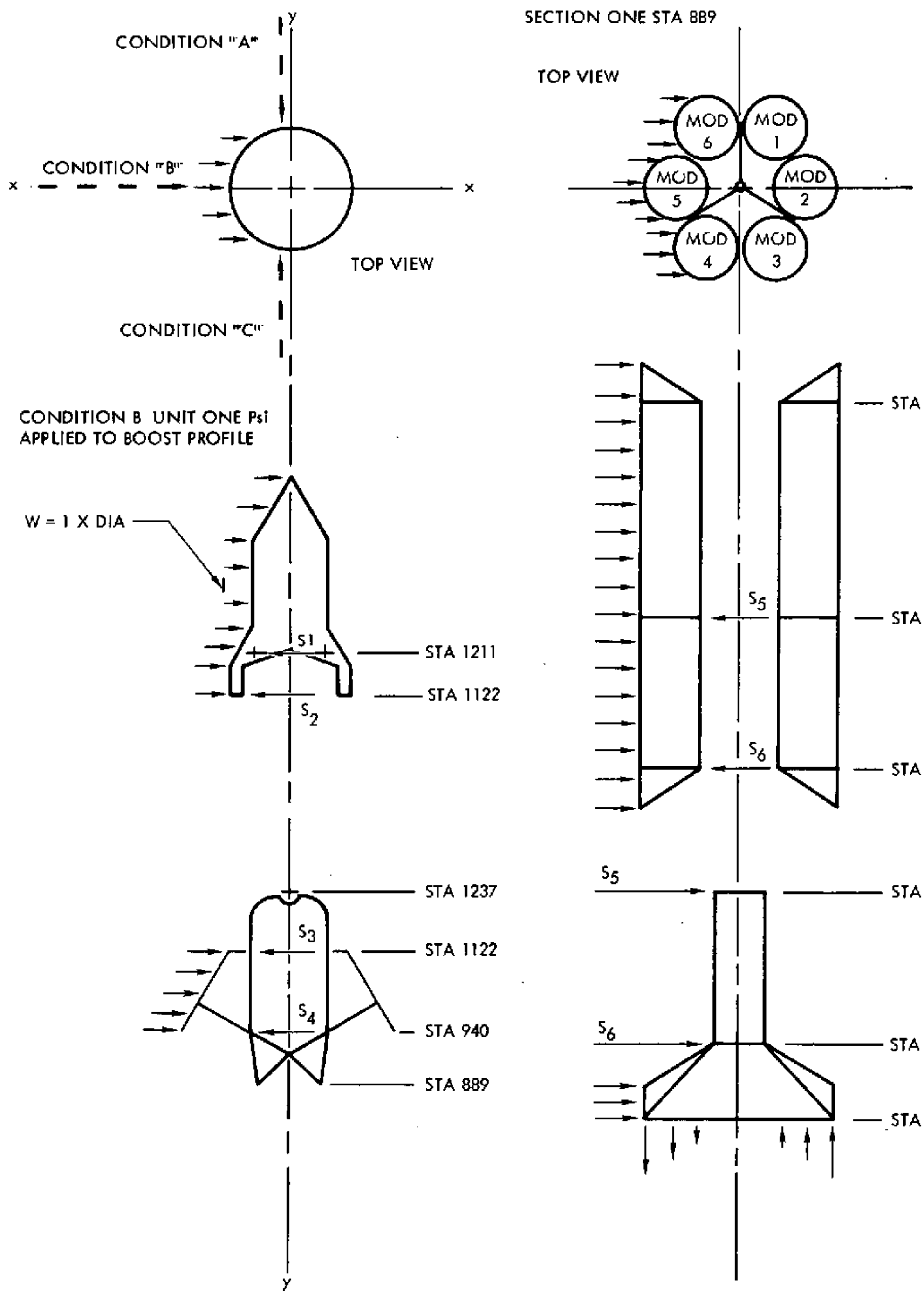


Figure 24. Unit One Psi Air Load Distribution

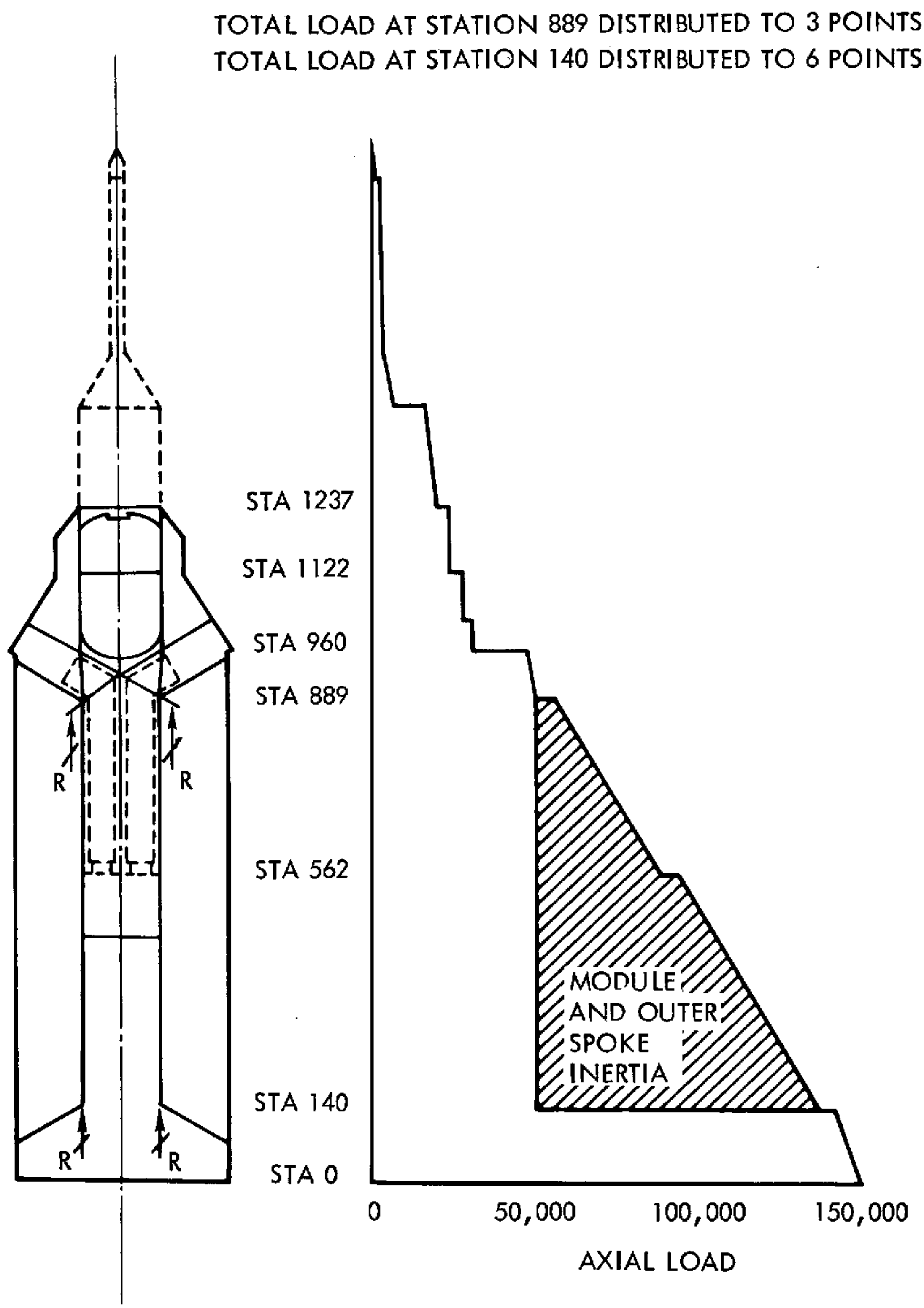


Figure 25. Unit One g Inertia Load Distribution

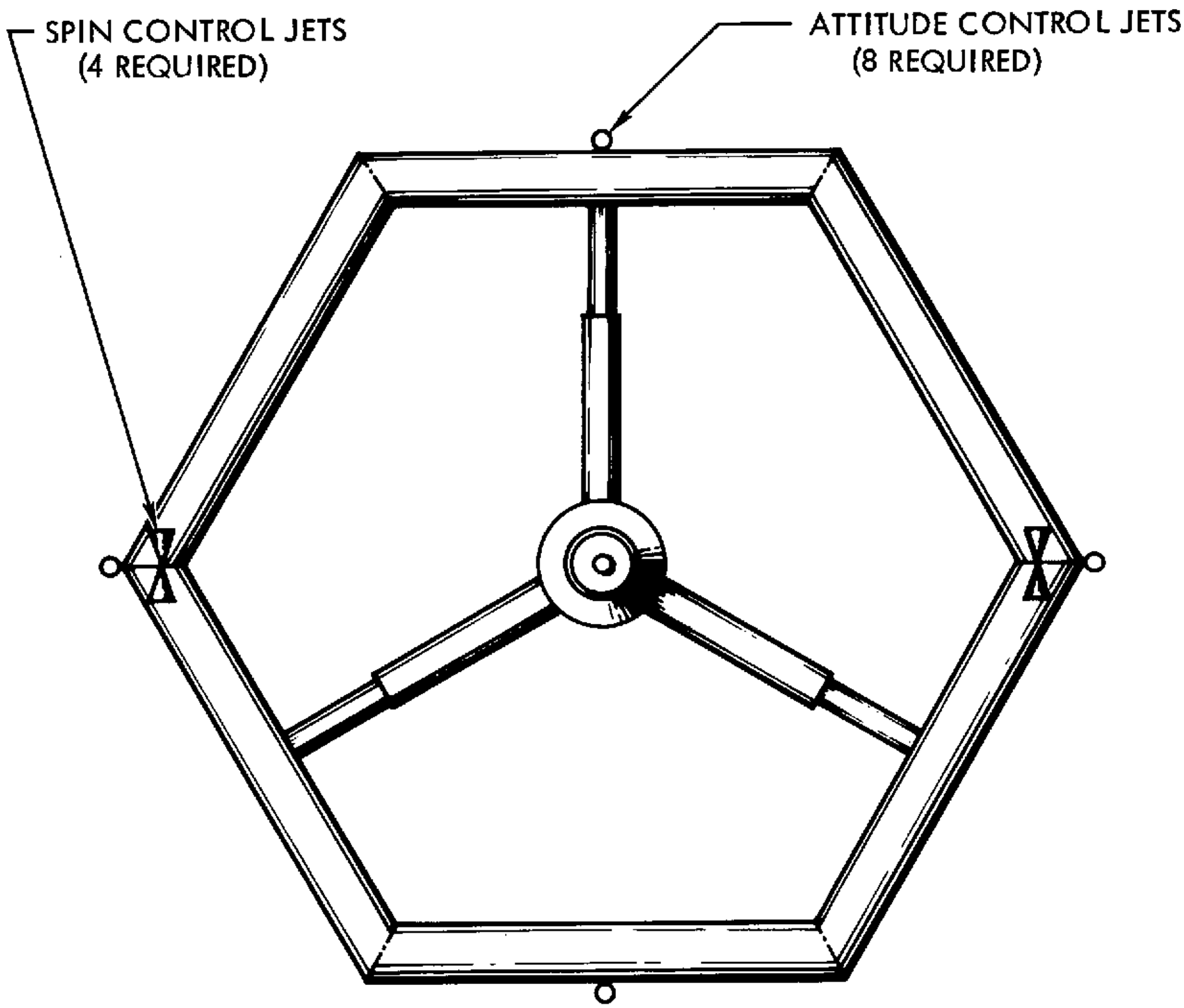


Figure 26. Control Force Location

Table 4. Unit and Actual Air Load Summary (Ultimate)

Condition		Station 1237	1122	1122	960	454	140
		LOAD (kips)					
		S_1	S_2	S_3	S_4	S_5	S_6
Unit Load on Profile for Condition 1 and 3 p = 1.00	A	76.0	-16.5	21.75	26.85	411.0	71.2
	B	76.0	-16.5	21.75	26.85	395.0	68.5
	C	76.0	-16.5	21.75	26.85	411.0	71.2
Unit Load on Cone for Condition 2 p = 1.00	A	76.0	-16.5	21.75	26.85	0	0
	B	76.0	-16.5	21.75	26.85	0	0
	C	76.0	-16.5	21.75	26.85	0	0
Condition p_1 = 0.139	A	10.6	-2.29	3.01	3.73	57.0	9.9
	B	10.6	-2.29	3.01	3.73	55.0	9.52
	C	10.6	-2.29	3.01	3.73	57.0	9.9
Condition p_2 = 2.29	A	174.0	-37.8	49.8	61.6	0	0
	B	174.0	-37.8	49.8	61.6	0	0
	C	174.0	-37.8	49.8	61.6	0	0
Condition p_3 = 1.27	A	96.5	-21.0	27.5	34.1	522.0	90.4
	B	96.5	-21.0	27.5	34.1	501.0	87.0
	C	96.5	-21.0	27.5	34.1	522.0	90.4

Actual Air Loads

Condition 1, Prelaunch:

$$q = 13.3\left(\frac{1.5}{144}\right) = 0.139 \text{ psi (ultimate)}$$ Applied to Entire Profile

Condition 2, Maximum Dynamic Pressure:

$$L = C_{N\alpha}\alpha qS$$

$$L = 0.028\,(120)\,57.3\left(\frac{3.14\,(396)^2}{576}\right) = 165{,}000 \text{ lb}$$

$$q = \left(\frac{165{,}000}{108{,}100}\right) 1.5 = 2.29 \text{ psi (ultimate)}$$ Applied to Conical Segment (Above Station 960)

Condition 3, Maximum Wind:

$$q = 122\left(\frac{1.5}{144}\right) = 1.27 \text{ psi (ultimate)}$$ Applied to Entire Profile

Table 5. Unit and Actual Axial Load Summary (Ultimate)

Item	Estimated Weights (10^{-3} lb)		Unit Nx = 1g	Station Application	Condition 1 Nx = 3g	Condition 2 Nx = 2.745g	Condition 3 Nx = 2.25g	Condition 4 Nx = 6.225g
	Equipment	Structure						
Apollo Command Module and Launch System	21.87							
Dock and Fairing		2.60	24.47	1237	73.5	67.3	55.0	152.2
Instrumentation and Power	5.35							
Laboratory and Fairing		5.00	24.47/29.82	1122	73.5/89.5	67.3/82.0	55/67.1	152.2/186
Inner Spoke		7.95						
			29.82/42.77	960	89.5/128.3	82.0/117.6	67.1/96.4	186/267
Hub Lower Support		7.23						
Total Modules and Contents (81.0)			50.0/55.0	889	150/165	137.5/151	112.5/124	311/342
Upper Module and Contents	21.0	20.0						
Outer Spoke		5.0	91.0/96.0	562	273/288	250/264	205/216	566/598
Lower Module and Contents	20.0	20.0						
			136/141.6	140	408/425	374/388	306/318	847/883
Thrust Structure Cylinder		5.6						
Thrust Structure Cone		8.4	150.0	0	450	411	338	935

Table 6. Unit One g Inertia Load Distribution

Equipment	Structure	Distribution	
		Weight	1g
Apollo Command Module		16, 000	16, 000
Launch Escape System		5, 870	21, 870
	Dock and Fairing	2, 600	24, 470
Instrumentation		2, 500	
Power System		2, 850	29, 820
	Lab and Fairing	5, 000	
	Inner Spokes	7, 950	42, 770
	Hub Lower Support	7, 230	50, 000
Attitude Control Communications Environmental Controls Instrumentation Personnel Accommodations Power System		41, 000	
	Modules and Outer Spoke	45, 000	136, 000
	Lower Cylinder Fairing	5, 600	141, 600
	Lower Conical Fairing	8, 400	150, 000

The loading on the spokes was not critical during the boost condition. Axial loads due to its own inertia did not contribute significantly toward the design of the spokes. During the orbital flight condition, loads distributed to the spokes were obtained by the principles of consistent deformation. This analysis was accomplished by using a torus shape rim (Figure 27); however, the same principles will apply for a hexagonal rim. The axial load distribution to the spokes due to pressurization and centrifugal force are determined by considering the structural system to be in the elastic range. Since the inner spoke hinge attachment points are located at the central hub bulkhead, the radial displacement of the bulkhead, due to the spoke loading, becomes virtually insignificant. It is therefore neglected in the load analysis. The outer spoke hinge attachment points, however, are located at the inner rim of the torus where the radial deflections become significant. The deflections considered are the elongation of the spokes, expansion of the torus under pressure, centrifugal force, and the effects of bending on the torus due to the radially redundant spoke load. The basic equations applying to the deflection terms are enumerated pertaining to the spoke and torus in the following:

Spoke

1. Elongation due to pressure and centrifugal force
$$\delta = \frac{\sigma_m L}{E} = \frac{\left(\frac{pR}{2t}\right) L}{E}$$

2. Influence coefficient
$$\delta = \frac{PL}{AE}$$

Torus

1. Displacement due to pressure and centrifugal force
$$\delta = \frac{\sigma_n R}{E} + \frac{WR^2}{EA}$$

2. Bending influence coefficient
$$\delta = -\frac{PR^3}{2E}\ \frac{1}{\sin^2\theta}\left(\frac{\theta}{2} + \frac{\sin\theta\cos\theta}{2}\right) - \frac{1}{\theta}$$

Thus, the redundant axial load distributed to each spoke is determined by equating the summation of the radial deflection terms of the spoke to the summation of the radial displacement terms of the torus. The axial load to each spoke is 313 pounds (ultimate) and therefore has negligible effect on spoke stresses.

Shear and bending moment distribution to the spokes due to the angular and pitch accelerations from the attitude control jets (antisymmetrical loading conditions) are resolved by a similar method used in obtaining the axial load in the spoke. By equating the end rotation and the tangential spoke deflection to the angle of twist and circumferential displacement of the torus, the redundant end moment and shear load is determined for the angular

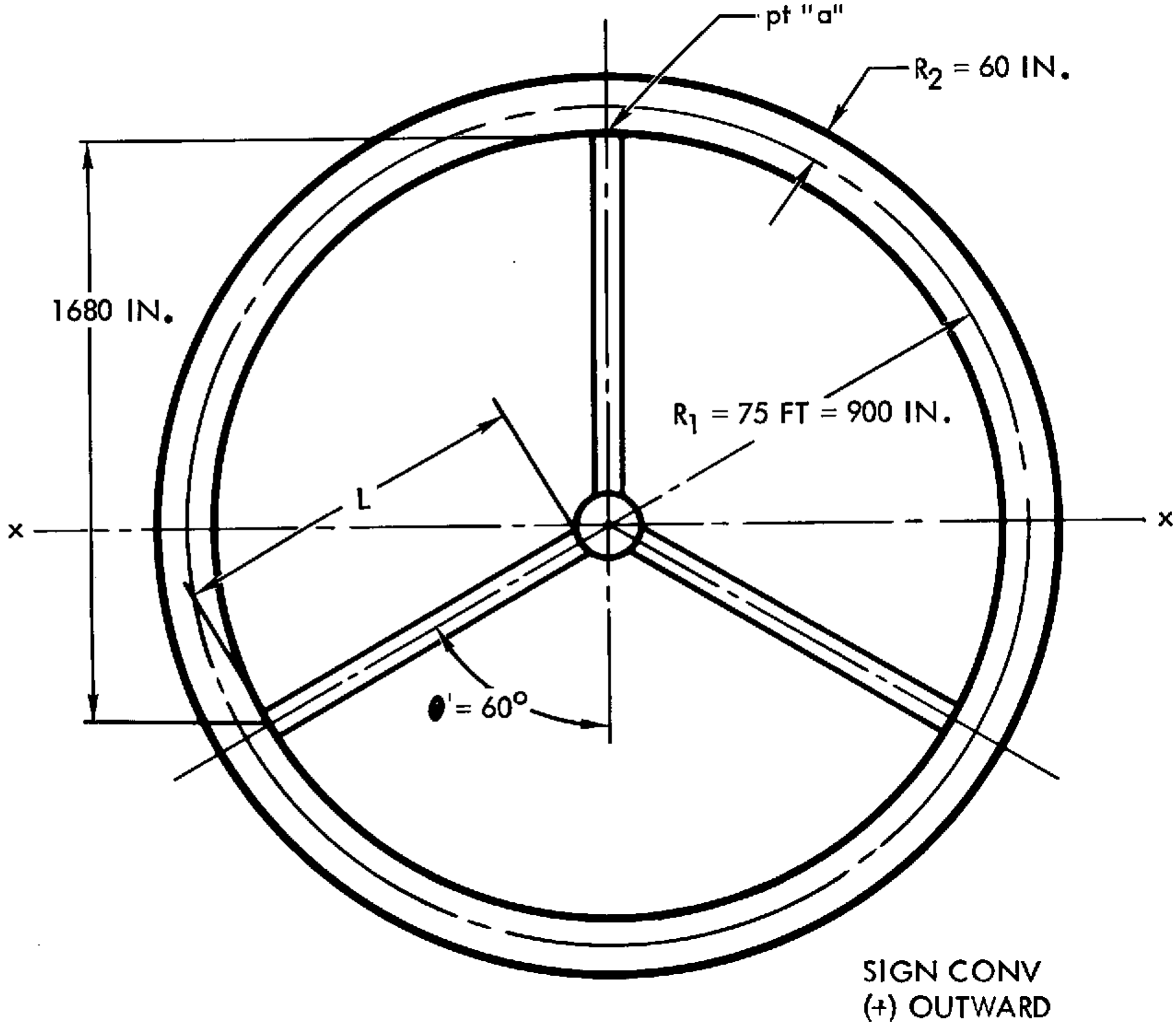


Figure 27. Station Geometry in Orbital Flight Conditions

acceleration loading condition. For pitch acceleration loading conditions, the internal loads are determined by equating rotations and deflections in the meridional direction to the torus. The loads resulting from this analysis have negligible effect on spoke stresses.

Structural Arrangement

A generalized description of the structural arrangement of the hub, modules, spokes, and thrust structure is presented in the following paragraphs.

Central Hub

The central hub consists of a rotating Apollo stowing carriage enclosing the dock and a fixed conical fairing enclosing the zero-g laboratory.

The skin fairing on the rotating stowing carriage is supported by transverse intercostals, beaming the surface pressure loading of the boost condition to 12 longitudinal beams. The longitudinal beams transmit the loading to the upper bulkhead and lower support frame where lateral support bearings are provided. Axial load from the structure above station 1237 (including Apollo command and service modules and launch escape system) is transmitted through the thrust bearing located at the skirt of the dock. The zero-g laboratory fairing skin is supported by stiffners and transverse frames transmitting the applied surface pressure loading to the upper support frame at station 1122 and to the conical bulkhead at the bottom of the zero-g laboratory. This bulkhead also supports the inner portion of the spoke. The entire central hub is supported by the modules at the upper cluster attachment fittings. Truss members forming three tripods distribute the fitting loads to the zero-g laboratory wall at the lower skirt. Support frames are provided to accommodate the induced kick loads.

The dock and zero-g laboratory walls are designed for meteoroid protection and are required to withstand internal pressure loading while the station is in orbit. The same type of construction is used on the bulkheads.

Modules

The modules are designed primarily for meteoroid protection and consist of six longitudional honeycomb panels joined by six longerons. Light frames are required, however, to improve local stability of the module wall for boost conditions.

During boost the modules are clustered to the thrust structure at the two sets of lower attachment fittings. Inertia loads on the outer portion of the spokes are distributed to the modules at station 562. The central hub loads induced at the upper cluster attachment fittings are transmitted to the thrust structure by individual bending in the modules.

Orbital flight body bending loads due to the attitude control jets do not contribute to the design of the modules. Pressurization and centrifugal force are the primary loads on the modules during orbital flight.

Spokes

The spokes are designed primarily for meteoroid protection and consist of honeycomb panels joined by longerons required due to panel size limitations. Light frames are used at the ends of the spokes to maintain the shell shape where it intersects the module and hub. Internal pressurization is the primary loading on the spokes.

Thrust Structure

The thrust structure, consisting of a cylinder, a cone frustrum and three support frames, is primarily designed by the boost loading conditions. The lower cluster fitting loads from the modules are applied to the support frames and transmitted by the shell to the booster.

Stress Analysis

A summary of the stress analysis which was conducted for the self-deploying space station is presented in the following paragraphs. For more detailed information, reference should be made to the supplementary stress analysis report.

The basic mechanical properties of 2014-T6 aluminum alloy material used in this study are listed in Table 7. The room temperature allowables were used for the analysis of orbital flight conditions and the 300 F allowables for the analysis of boost conditions.

Support frames are provided in the shell structures where concentrated loads are introduced. A parametric study of frames, including effects of shear resistant webs and honeycomb webs, was prepared in order to select the optimum design for the various loading range using the 2014-T6 material operating at 300 F. This study is summarized in Figure 27 and shows the allowable bending moment and shear load versus the frame cross section area and frame height.

The compression allowables of the shell discussed in the "Various Types of Construction" are used in determining the shell gauges when strength is the critical design criteria.

Table 7. Mechanical Properties of 2014-T6 Aluminum Alloy

Mechanical Properties		Temperature: *300 F	Temperature: Room Temperature
F_{tu}	Long.	51.0	60.0
	Trans.	46.0	54.0
F_{ty}	Long.	45.0	53.0
	Trans.	40.0	48.0
F_{cy}	Long.	47.0	55.0
	Trans.	45.0	53.0
F_{su}		39.0	39.0
μ		0.33	0.33
E		10.1×10^6	10.6×10^6

Note: Minimum Guaranteed Values
All Stresses KSI
Long. - Longitudinal Grain
Trans. - Transverse Grain
* 1/2 hr exposure time

Methods Used in Sizing Members

Sizing of the frame members consists of obtaining the bending moments, axial loads, and shear loads and selecting the optimum configuration from Figure 28. Conventional beam theory is used in determining reactions, bending moments, and shear loads. For continuous frames, the ring report, Reference 9, is used in obtaining the bending moments, shear loads, and axial loads.

For bending moments applied on circular cylinders, the wall loading is obtained in the following:

$$w_b = \frac{M}{\pi R^2}$$

Axial loading on the circular cylinder is obtained from

$$w_a = \frac{P}{2\pi R}$$

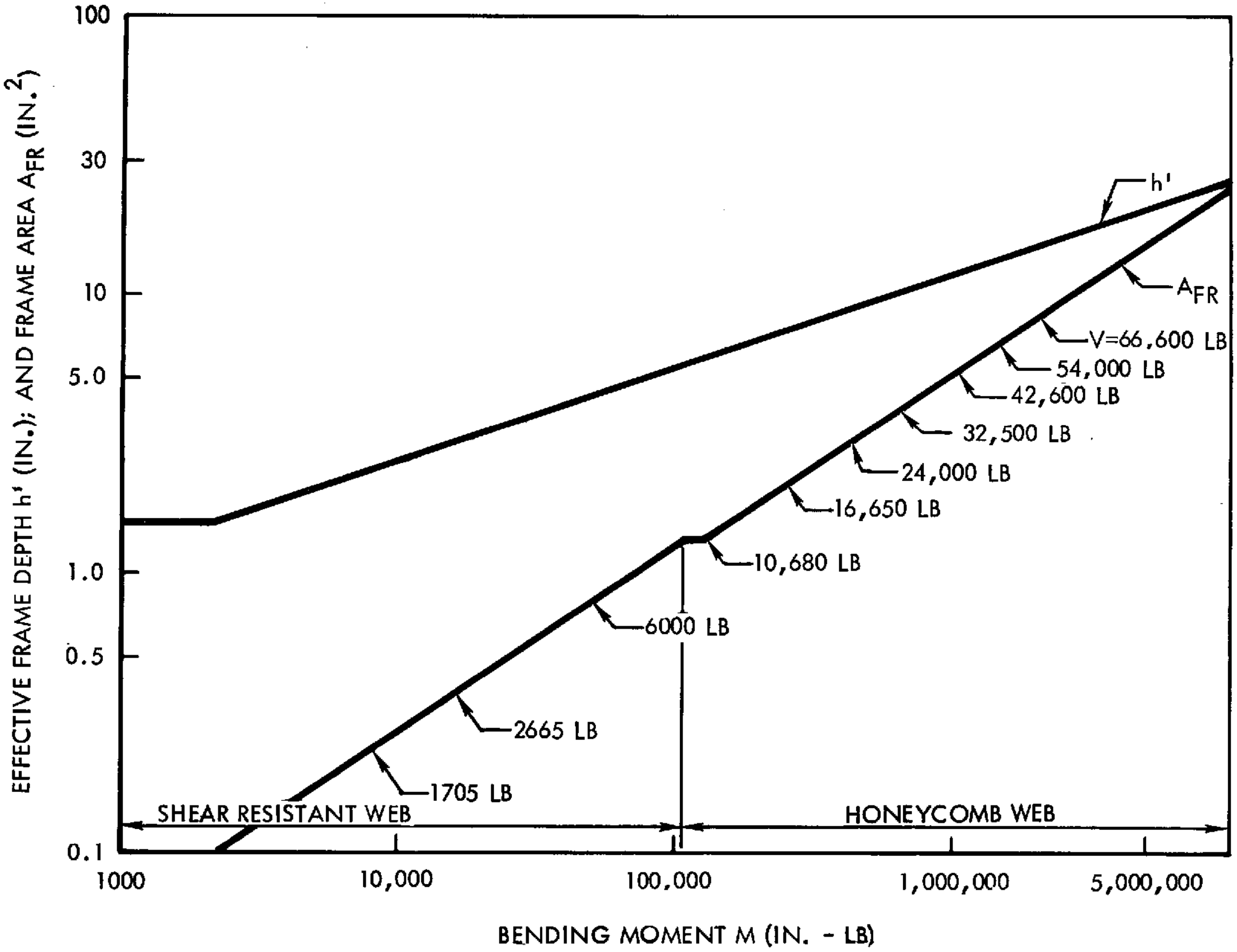


Figure 28. Frame Sizing - 2014 T-6 at 300 F

Both general and local instability modes of failure are checked for the column members.

Conventional pressure vessel analysis methods are used in obtaining the meridional and hoop tension loading in the shell due to pressurization. A detailed analysis of the joint discontinuity stresses at a typical bulkhead of the modules shows a small increase in the membrane stresses.

Summary

A summary of the stress analysis of the final configuration is presented in Table 8. This summary is divided into four sections; the central hub, modules, spokes, and thrust structure. These sections are further divided into major components and then into structural members. The structural member sizes are given in terms of cross section area for frames and longerons and for thickness for skins.

In most cases, the boost condition is the critical design criterion. The orbital flight conditions and member margins of safety are also shown for comparison.

Table 8. Final Configuration Stress Analysis Summary

Component	Member	Size		Boost		Orbit Flight	
		Frame or Long. Area	Skin t	Cond.	M. S.	Cond.	M. S.
CENTRAL HUB							
Dock Fairing	Intercostal Beams	0.36	-	2	0	(Not critical)	-
	Skin	-	0.050	2	0.10		-
	Stringer	0.351	-	2	0.10		-
	Support Beams (Sect. A)	0.600	-	2	0		-
	Support Beams (Sect. B)	1.40	-	2	0		-
	Bulkhead Edge Member	1.20	-	2	0		-
	Bulkhead Web	-	0.024	2	0		-
	Lower Support Frame	2.0	-	2	0		-
Dock Shell	Wall	-	0.076	2	0.53	Pressure	0.32
	Dome	-	0.076	2	0.11	Pressure	0.32
Lab Fairing	Skin	-	0.050	2	0.05	(Not critical)	-
	Stringers	0.351	-	2	0.05		-
	Upper Support Frame	1.60	-	2	0		-
	Conical Bulkhead	-	0.076	2	0.24		-
Lab. Shell	Wall	-	0.076	2	0.08	Pressure	0.32
	Dome	-	0.076	2	-	Pressure	0.32
Truss	Center Struts	1.81	-	2	0	(Not critical)	-
	Outer Struts	6.25	-	2	0		-
	Support Frame	1.60	-	2	0		-
SPOKE							
Shell	Wall	-	0.076	(Not critical)	-	Pressure	1.40
	Support Frame	0.40	-		-	Pressure	-
MODULE							
Shell	Wall	-	0.076	2	0.14	Press & centrifugal force	0.70
	Dome	-	0.076	(Not critical)			0.70
Support Structure	Frame Sta. 889	4.00	-	3	0	(Not critical)	-
	Frame Sta. 450	7.20	-	3	0		-
	Frame Sta. 140	4.80	-	3	0		-
THRUST STRUCTURE							
Shell	Cylinder (Top)	-	*0.214	3	0	(Not critical)	-
	Cylinder (Bottom)	-	*0.390	3	0		-
	Cone Frustrum (Top)	-	*0.647	3	0		-
	Cone Frustrum (Bottom)	-	*0.096	3	0		-
Support Structure	Frame Sta. 450	15.2	-	3	0	(Not critical)	-
	Frame Sta. 140	41.0	-	2	0		-
	Frame Sta. 0	7.6	-	3	0		-

*NOTE: Equivalent thickness of skin and stringer

SPACE ENVIRONMENT

There are a great number of factors associated with the space environment, the most important of which are pressure, temperature, natural radiation, and the presence of meteoroid particles. In a manned vehicle whose mission duration is relatively short, pressure and temperature are the most important factors to consider in design. The radiation dosage received by the crew members is a function of exposure time; and, similarly, the probability of meteoroid damage is a function of mission duration. Thus, these factors are relatively unimportant for short missions but become primary considerations in designing a structure for missions of long duration.

METEOROID ENVIRONMENT

All space vehicles will be subjected to many collisions with meteroids when they are out of the earth's atmosphere. Effects of these collisions may take the form of slow surface deterioration, pitting of exposed areas, or actual penetration of the vehicle. In an extreme case, penetration may destroy instrumentation within the vehicle or kill the occupant. Generally, however, the chance of encounters with such large meteoroids is remote. The major damage to man or equipment within a space vehicle involves the change in environment by escape of the vehicle atmosphere. Meteroids may range in size from a radius of 10^{-4} centimeters to a radius of several meters. The density of meteroids may possibly vary from that of iron for the smallest particles to as low as 0.05 grams per cubic centimeter for the largest particles.

The techniques for projection of hypervelocity particles and for observing the effects of impact have improved considerably in recent years. The variations in masses of projectiles used in various tests cover the range of primary concern in space travel, but the velocities only extend into the lower range of interest. The laboratory tests of hypervelocity penetration have led to many formulas, both empirical and theoretical. The formula developed by NASA Ames Research Center agrees well with test data obtained using projectile velocities of 20,000 feet per second and higher. This formula, used in all of the NAA analyses, may be expressed as follows:

$$s_1 = 5.66\, m^{1/3} \left(\rho_p\right)^{1/3} \left(\frac{V}{\rho_t C}\right)^{2/3}$$

where s_1 is the thickness in centimeters of a single-sheet structure that will just resist penetration, ρ_p is the density of the meteoroid in grams per cubic centimeter, ρ_t is the density of the structure material in grams per cubic centimeter, V is the velocity of impact in feet per second, and C is the bar speed of sound in the structure material in feet per second.

Preliminary studies have indicated that rigid space structures can provide adequate meteoroid protection and that flexible structures dense enough to provide adequate meteoroid penetration resistance cannot readily be folded into compact launch configurations. It has been found that a single-sheet structure that has a large exposed surface area of either inflatable or rigid material is not practical for a long exposure time. However, a satisfactory structure is obtained by dividing the single-sheet thickness into multiple-sheets with spacing and fillers between them.

During the study, various multisheet rigid and nonrigid structures were analyzed for penetration resistance to obtain the optimum structure for the mission.

NATURAL RADIATION ENVIRONMENT

During certain missions, limited radiation damage might be expected to the structure, nonstructural materials, certain electronic parts, and vehicle systems as well as to the crew members on space stations. All results presented in this report are based entirely on presently available data; recent Explorer XII and Injun I findings (not yet available) may greatly affect many of these results.

The nuclear environments encountered during a mission may consist of the ionosphere, the inner Van Allen belt, and the outer Van Allen belt and may be influenced by solar cosmic rays and galactic cosmic rays. The magnitude of the possible effects and length of exposure to cosmic rays, solar cosmic ray events, and leakage of electrons from the outer Van Allen belt requires that protection of the crew and other systems be considered in the structural design. Possible solutions to the radiation problems are shielding (electrostatic, absorbing shields, scatter shields), solar storm prediction, choice of trajectories that utilize the shielding provided by the earth's magnetic field, and return of the personnel to earth. Further studies will define the need for these protective measures. No shielding appears to be necessary for orbits below a 400-nautical-mile altitude if the inclination of the orbit to the geomagnetic equator is below approximately 40 degrees (geocentric inclination = 30 degrees). However, above 40 degrees geomagnetic latitude, shielding, solar storm prediction, etc., will be required or the allowable duration of the men in orbit will have to be reduced. Choice of materials to be used in a manned space vehicle depends not only on their structural properties before and after exposure to radiation, but also on their secondary emissivity. Aluminum appears to be the best structural material because its mechanical properties are not appreciably affected and because its secondary emissivity cross section is small. Little is known at this time about the effect of very high energy particles on structural metals.

The radiation environment was described in more detail in the midterm report and will not be repeated herein.

STABILIZATION AND CONTROL

The stabilization and control study effort was directed toward the preliminary analysis of possible problem areas. In contrast to detailed studies, this approach allowed a broader scope of investigation.

The studies which were conducted may be classified into two general areas: disturbances that will act on the vehicle and systems capable of counteracting these disturbances and maintaining the desired attitude orientation. The disturbances studied include crew motion, docking impact, gravity gradient, meteorite impact, mass unbalance, circulating fluids, and rotating machinery. A reaction-jet attitude reorientation system was devised, and the required weight of propellant was estimated. Two wobble damping systems investigated were a reaction-jet system and a momentum wheel precession system.

There appears to be no question of feasibility of the stabilization and control system. The incorporation of a mass conservative wobble damper would minimize the oscillatory accelerations imposed on the crew without a major weight or power penalty on the vehicle. Furthermore, the estimated weight of reaction jet propellant required to maintain a solar reference does not appear excessive.

Recommendations, based on the effort to date, concerning the application of future effort in the stabilization and control area have been included at the end of the report.

A summary of the general terminology used throughout the stabilization and control section of the report is presented in the following listings:

Definition of Terms

a	Linear acceleration (ft/sec^2)
D	Diameter (ft)
H	Angular momentum (ft-lb-sec)
I	Moment of inertia (sl-ft^2)
I_{sp}	Specific impulse (lb-sec/lb)

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I_{xy} Product of inertia with respect to x and y axes (typical) (sl-ft^2)

J Angular impulse (ft-lb-sec)

K Control system gain

l Reaction jet lever arm (ft)

m Angular attitude error about the z body axis (rad)

n Angular attitude error about the y body axis (rad)

N Angular rotation rate (rpm)

p Angular rotation rate about the vehicle x body axis (rad/sec)

P Power (watts)

q Angular rotation rate about the vehicle y body axis (rad/sec)

r Angular rotation rate about the vehicle z body axis (rad/sec)

R Radius (ft)

s Independent variable of the La Place transform

S Allowable stress (lb/ft^2)

t Time (sec)

u Angle between the orbital radius vector and the line of nodes (rad)

U(t) Unit step function

W Weight (lb)

Definition of Greek Symbols

α Angle between the vehicle x body axis and the normal to the orbit plane (rad)

β Angle between the vehicle spin axis and the projection of a vector normal to the orbit plane on the plane of the ecliptic (rad)

γ Wheel "gimbal" displacements about vehicle axis (rad)

$\delta(t)$ Unit impulse function

η Sun sensor optical constant

θ 1. Euler angle, about y body axis (rad)
2. Total attitude error, angle between the vehicle x body axis and the sun line (rad)

θ_w Angle subtended by thickness of wheel (rad)

λ $\dfrac{(I_x - I_y)}{(I_y)}$

μ Product of the earth's mass and the universal gravitational constant (ft^3/sec^2)

ν Number of spin periods

ρ Density (sl/ft^3)

τ Time constant (sec)

ϕ Euler angle, about the x body axis (rad)

ψ Euler angle, about the z body axis (rad)

ω General angular rate (rad/sec)

ω_n System natural frequency (rad/sec)

Definition of Subscripts

c Control

db Dead band

e Effective

g Due to gravity

I Inertial

j Jets

l Large

o Initial

p Propellant

s Small

sl Sun line

ss Steady State

x About or along the x axis

y About or along the y axis

z About or along the z axis

Definition of Superscripts

• Denotes derivative with respect to time

EQUATIONS OF MOTION

The following section is a development of the equations of motion used in the subsequent analyses. Mathematical expressions are developed for the sun sensor signals as functions of the vehicle body rates. Precession wheel reactive torques are also evolved. In addition, equations defining the inertial motion of the space station spin axis have been included. The equations developed are written in terms of an orthogonal, body axis coordinate system with the origin at the mass center of the vehicle and the x body axis colinear with the vehicle spin axis.

Vehicle Equations of Motion

The components of vehicle momentum are as follows:

$$H_x = I_x p - I_{xy} q - I_{xz} r$$

$$H_y = I_y q - I_{xy} p - I_{yz} r$$

$$H_z = I_z r - I_{xz} p - I_{yz} q$$

Then the vehicle rates in terms of the applied torques are

$$T_i = \left.\frac{d\bar{H}}{dt}\right|_i = \frac{\partial H_i}{\partial t} + (\bar{\Omega} \times \bar{H})_i$$

$$T_x = I_x\dot{p} + p\dot{I}_x - I_{xy}\dot{q} - q\dot{I}_{xy} - I_{xz}\dot{r} - r\dot{I}_{xz} + q(I_z r - I_{xz}p - I_{yz}q)$$
$$-r(I_y q - I_{xy}p - I_{yz}r)$$

$$T_y = I_r\dot{q} + q\dot{I}_y - I_{xy}\dot{p} - p\dot{I}_{xy} - I_{yz}\dot{r} - r\dot{I}_{yz} + r(I_x p - I_{xy}q - I_{xz}r)$$
$$-p(I_z r - I_{xz}p - I_{yz}q)$$

$$T_z = I_z\dot{r} + r\dot{I}_z - I_{xz}\dot{p} - p\dot{I}_{xz} - I_{yz}\dot{q} - q\dot{I}_{yz} + p(I_y q - I_{xy}p - I_{yz}r)$$
$$-q(I_x p - I_{xy}q - I_{xz}r)$$

Considering small signal perturbations of p, q, and r and a stepwise product of inertia change I_{xy} U(t), the above equations can be La Place transformed as follows:

$$T_x = I_x sp - I_{xy}sq + I_{xy}\,p_o r$$

$$T_y = -\,p_o I_{xy} - I_{xy}sp - I_y sq + p_o \lambda I_y r$$

$$T_z = -\frac{p_o^{\,2} I_{xy}}{s} - 2p_o I_{xy}p - p_o\lambda I_y q + I_y sr$$

In developing the foregoing equations, we have assumed the initial conditions q(o+) and r(o+) to be equal to zero. There is no loss of generality to assume only the I_{xy} U(t) disturbance. By considering small perturbations in the rates, we are automatically linearizing the equations so that the effects of the step changes in the other product of inertia terms can be superimposed later. The solution of these equations in the time domain for p is

$$p(t) = \frac{p_o I_{xy}^2 (2I_x - I_y)}{\left[I_x(I_x - I_y)^2 + I_{xy}^2(3I_x - I_y)\right]} \left\{ 1 - \left(1 - \frac{\left[I_x(I_x - I_y)^2 + I_{xy}^2(3I_x - I_y)\right]}{\left[I_x I_y - I_{xy}^2\right]\left[2I_x - I_y\right]}\right) \cos \omega_1 t \right\}$$

where ω_1 = wobble frequency

Substituting the values of moments of inertia associated with the space station configuration under study and assuming a large product of inertia, $I_{xy} = 0.1 \times 10^6$:

$$p(t) = p_o - \frac{(0.1)^2 (2 \times 15 - 10) p_o}{\left[15(15 - 10)^2 + (0.1)^2 (3 \times 15 - 10)\right]} \left\{ 1 - \left(1 - \frac{15(15 - 10)^2 + (0.1)^2 (3 \times 15 - 10)}{\left[15 \times 10 - (0.1)^2\right]\left[2 \times 15 - 10\right]}\right) \cos \omega_1 t \right\}$$

$$= p_o \left\{1 - 0.0005(1 - 1.125 \cos \omega_1 t)\right\} \equiv p_o$$

Hence, all further analysis can assume essentially no perturbation in p. Solving the original equations for q and r in terms of products of inertia steps I_{xy} U(t), I_{xz} U(t) and I_{yz} U(t) and torques T_y, T_{yc}, T_z and T_{zc} we obtain:

$$q = \frac{\left(T_{yc} + T_y + I_{xy} p_o - I_{xz} p_o^2/s\right) I_z \left(s - \frac{p_o I_{yz}}{I_z}\right)}{\left(I_y^2 - I_{yz}^2\right)\left(s^2 + \frac{p_o^2\left[\lambda^2 I_y^2 - I_{yz}^2\right]}{\left[I_y^2 - I_{yz}^2\right]}\right)}$$

$$+ \frac{\left(T_{zc} + T_z + I_{xz} p_o + I_{xy} p_o^2/s\right) I_{yz} \left(s - \frac{\lambda I_z p_o}{I_{yz}}\right)}{\left(I_y^2 - I_{yz}^2\right)\left(s^2 + \frac{p_o^2\left[\lambda^2 I_y^2 - I_{yz}^2\right]}{\left[I_y^2 - I_{yz}^2\right]}\right)}$$

$$r = \frac{\left(T_{yc} + T_y + I_{xy} p_o - I_{xz} p_o^2/s\right) I_{yz} \left(s + \frac{\lambda I_y p_o}{I_{yz}}\right)}{\left(I_y^2 - I_{yz}^2\right)\left(s^2 + \frac{p_o^2\left[\lambda^2 I_y^2 - I_{yz}^2\right]}{\left[I_y^2 - I_{yz}^2\right]}\right)}$$

$$+ \frac{\left(T_{zc} + T_z + I_{xz} p_o + I_{xy} p_o^2/s\right) I_y \left(s + \frac{\lambda I_y p_o}{I_y}\right)}{\left(I_y^2 - I_{yz}^2\right)\left(s^2 + \frac{p_o^2\left[\lambda^2 I_y^2 - I_{yz}^2\right]}{\left[I_y^2 - I_{yz}^2\right]}\right)}$$

Sun Sensor Signals

The sun sensor outputs m and n, along the vehicle's y and z axes respectively, can be related to the body rates through the Euler angles. Consider the rotational sequence to be ψ, θ and ϕ about the z, y, x body axes respectively as illustrated in Figure 29. Then

$$m = (\cos\psi \sin\theta \sin\phi - \sin\psi \cos\phi)\,\eta \cong (\theta \sin\phi - \psi \cos\phi)\,\eta$$

$$n = (\cos\psi \sin\theta \cos\phi + \sin\psi \sin\phi)\,\eta \cong (\theta \cos\phi + \psi \sin\phi)\,\eta$$

$$\dot{\phi} = p + \tan\theta\,(q \sin\phi + r \cos\phi) \cong p_o; \therefore \phi = pt$$

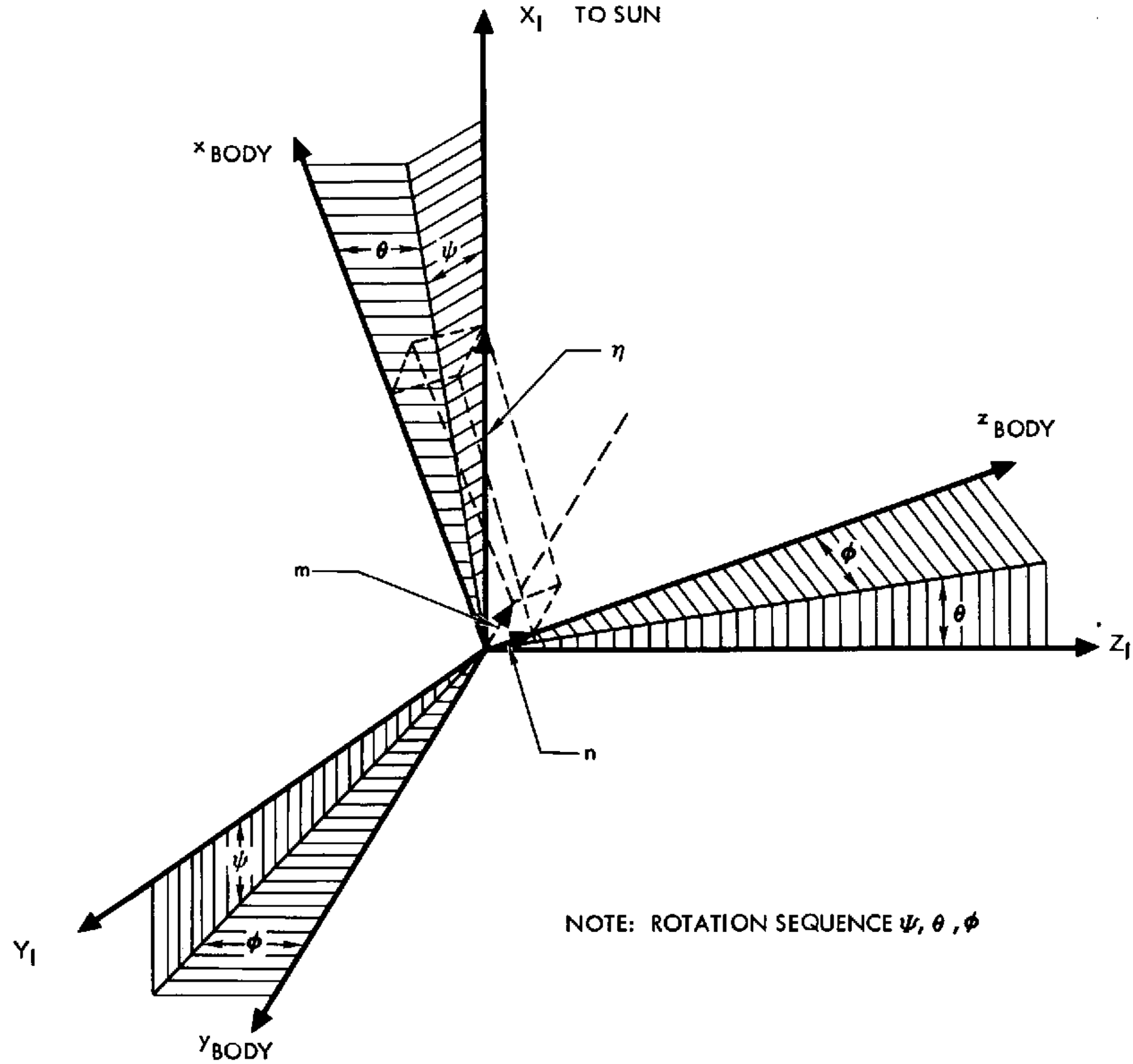


Figure 29. Body Axis Orientation With Respect to the Inertial Reference

$$\dot{\psi} = q \frac{\sin \phi}{\cos \theta} + r \frac{\cos \phi}{\cos \theta} \cong q \sin p_o t + r \cos p_o t$$

$$\dot{\theta} = q \cos \phi - r \sin \phi \cong q \cos p_o t - r \sin p_o t$$

where: θ and ψ are small angles so that

$$\cos \theta \cong \cos \psi \cong 1; \sin \theta \cong \theta; \sin \psi \cong \psi$$

$$q \tan \theta \sin p_o t << p_o$$

Through the use of complex convolution, these relationships yield the following s domain solutions for m and n:

$$m(s) = \frac{\eta}{\left(s^2 + p_o^2\right)}\left[p_o q - sr\right]$$

$$n(s) = \frac{\eta}{\left(s^2 + p_o^2\right)}\left[sq + p_o r\right]$$

These equations can be manipulated to give:

$$sm(s) = - r + p_o n$$

$$sn(s) = q - p_o m$$

Inertial Motion of Spin Axis

The traces of a unit vector along the vehicle spin axis (Figures 30 to 34) were obtained by substituting the solutions for q and r into the Euler equations for $\dot{\theta}$ and $\dot{\psi}$ and then integrating to obtain θ and ψ. If the body axes are assumed to be aligned with the inertial axes at t = 0 then ψ and $-\theta$ correspond closely to the positions of the tip of the unit vector along the inertially fixed y and z axes respectively. Thus, the inertial equations of motion for an impulsively applied torque $T_z \delta(t)$ with an undamped system (Figure 30) are as follows:

$$y_I = \frac{T_z}{I_y p_o} \left\{ \frac{\sin (1 + \lambda) p_o t}{(1 + \lambda)} \right\}$$

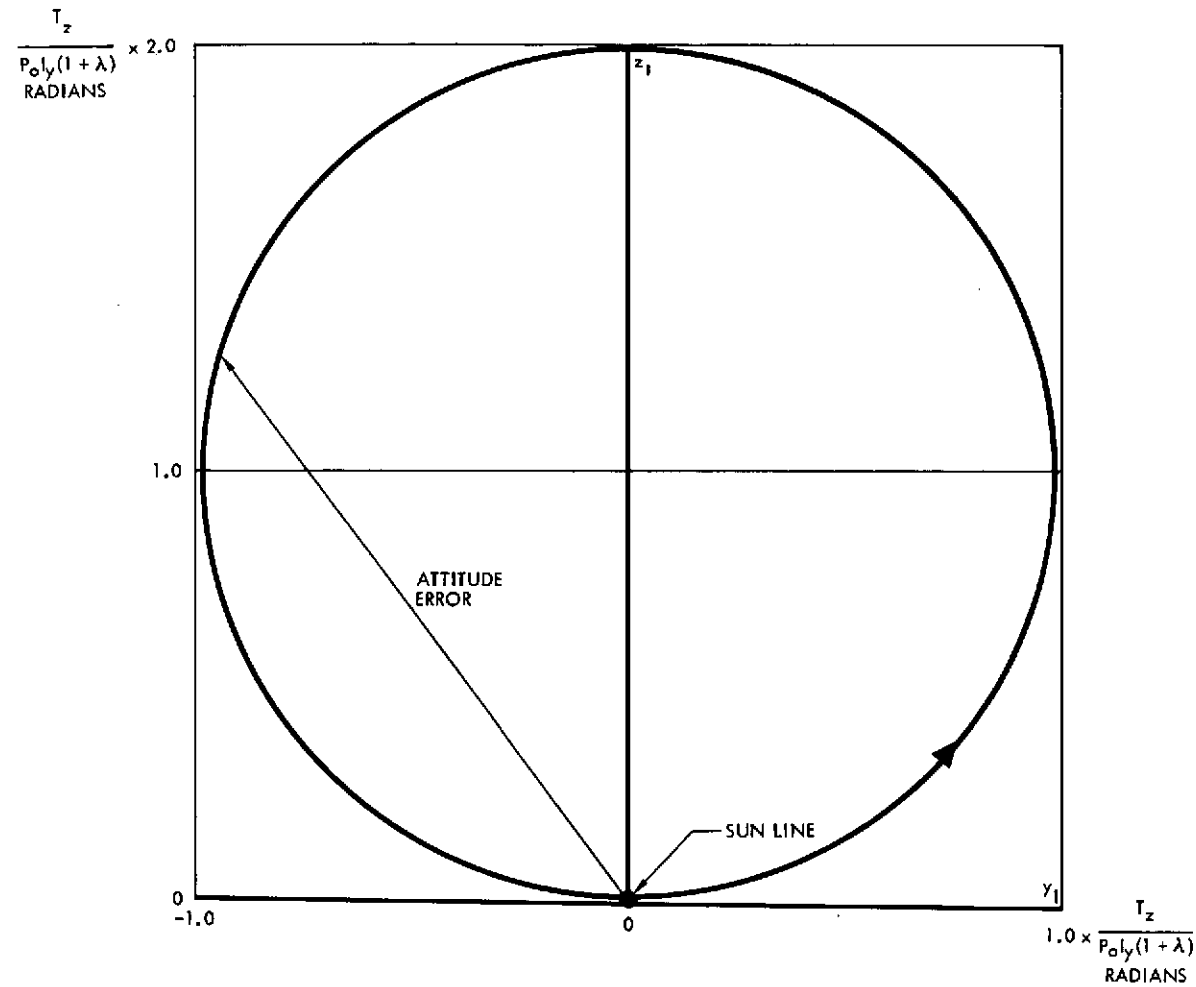


Figure 30. Inertial Motion of the Tip of a Unit Vector Along the Vehicle Spin Axis for Impulsive Torque Disturbance Undamped System

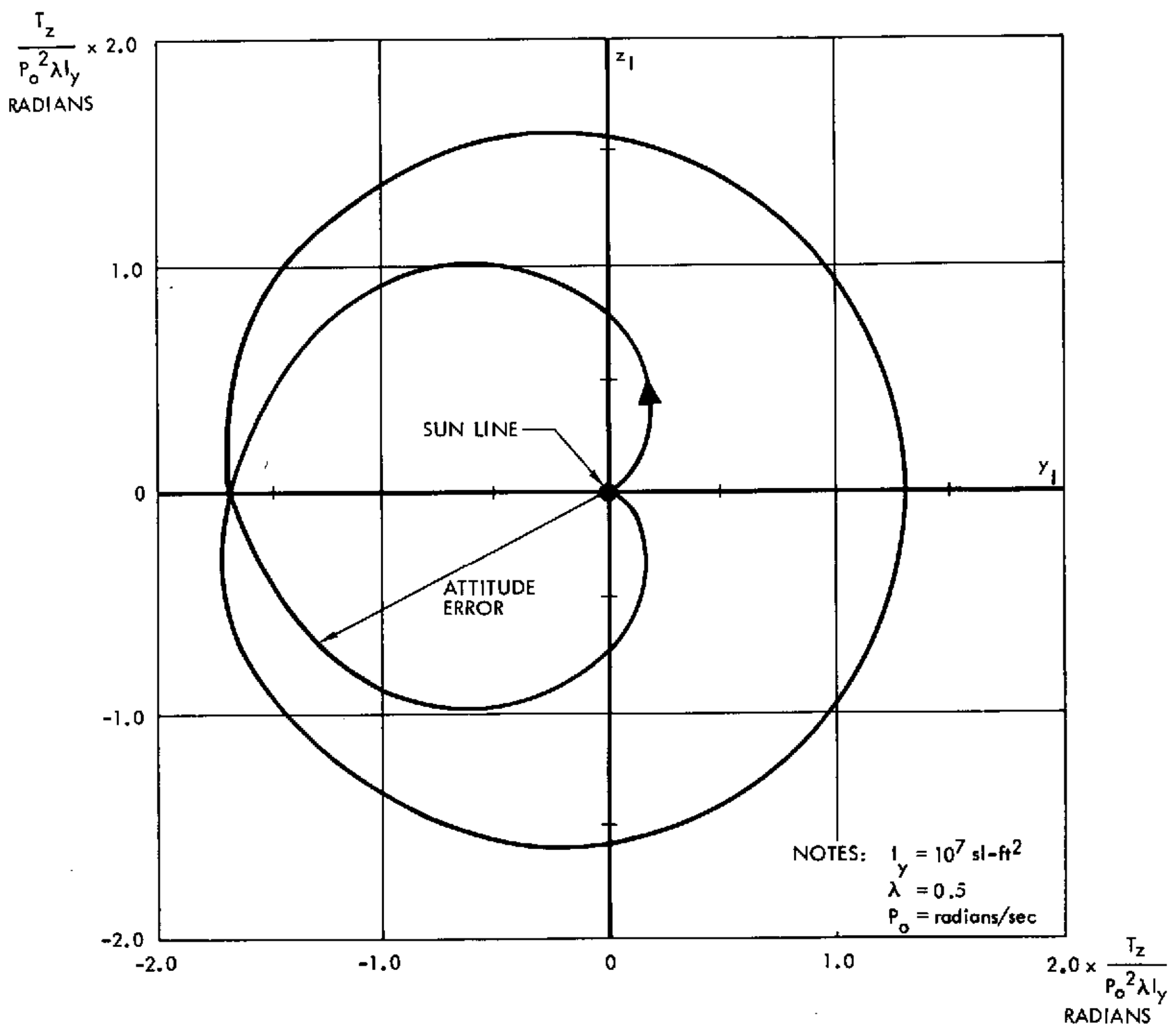


Figure 31. Inertial Motion of the Tip of a Unit Vector Along the Vehicle Spin Axis for Step-Torque Disturbance $[T_z\ U(t)]$ - Undamped System

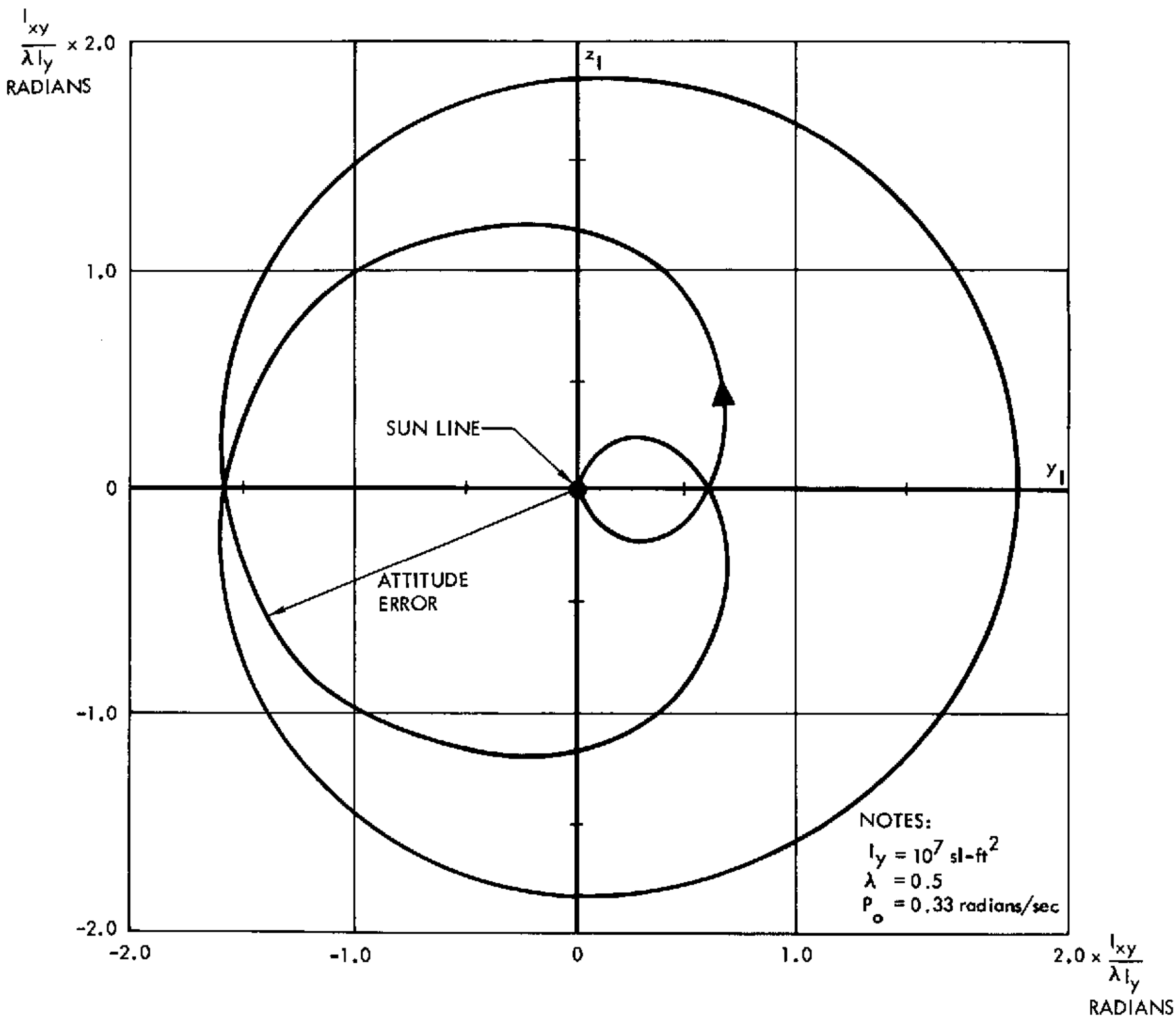


Figure 32. Inertial Motion of the Tip of a Unit Vector Along the Vehicle Spin Axis for Product-of-Inertia Disturbance $[I_{xy}\ U(t)]$ - Undamped System

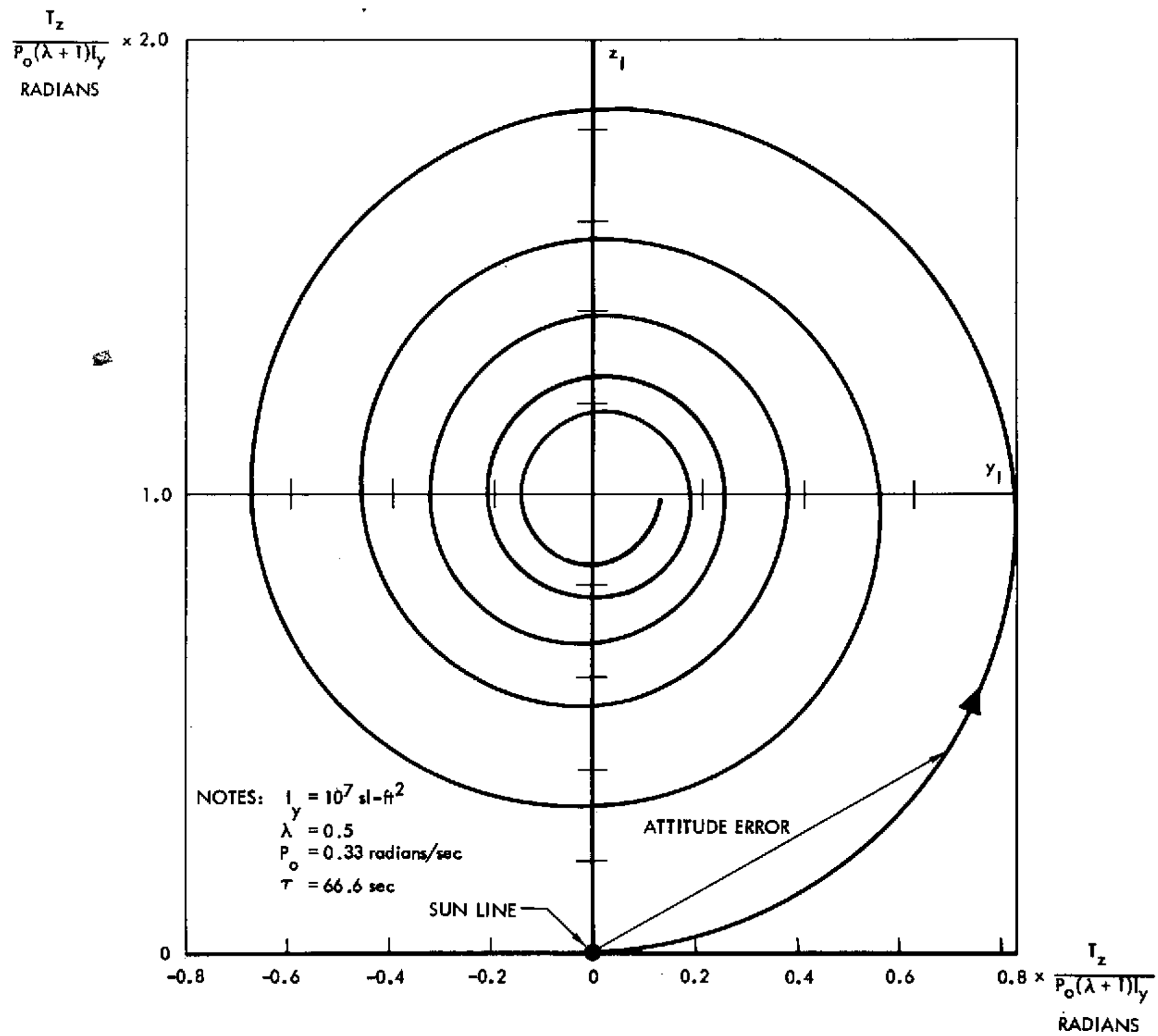


Figure 33. Inertial Motion of the Tip of a Unit Vector Along the Vehicle Spin Axis for Impulsive-Torque Disturbance $[T_z \, \delta(t)]$ -Rate Controlled System

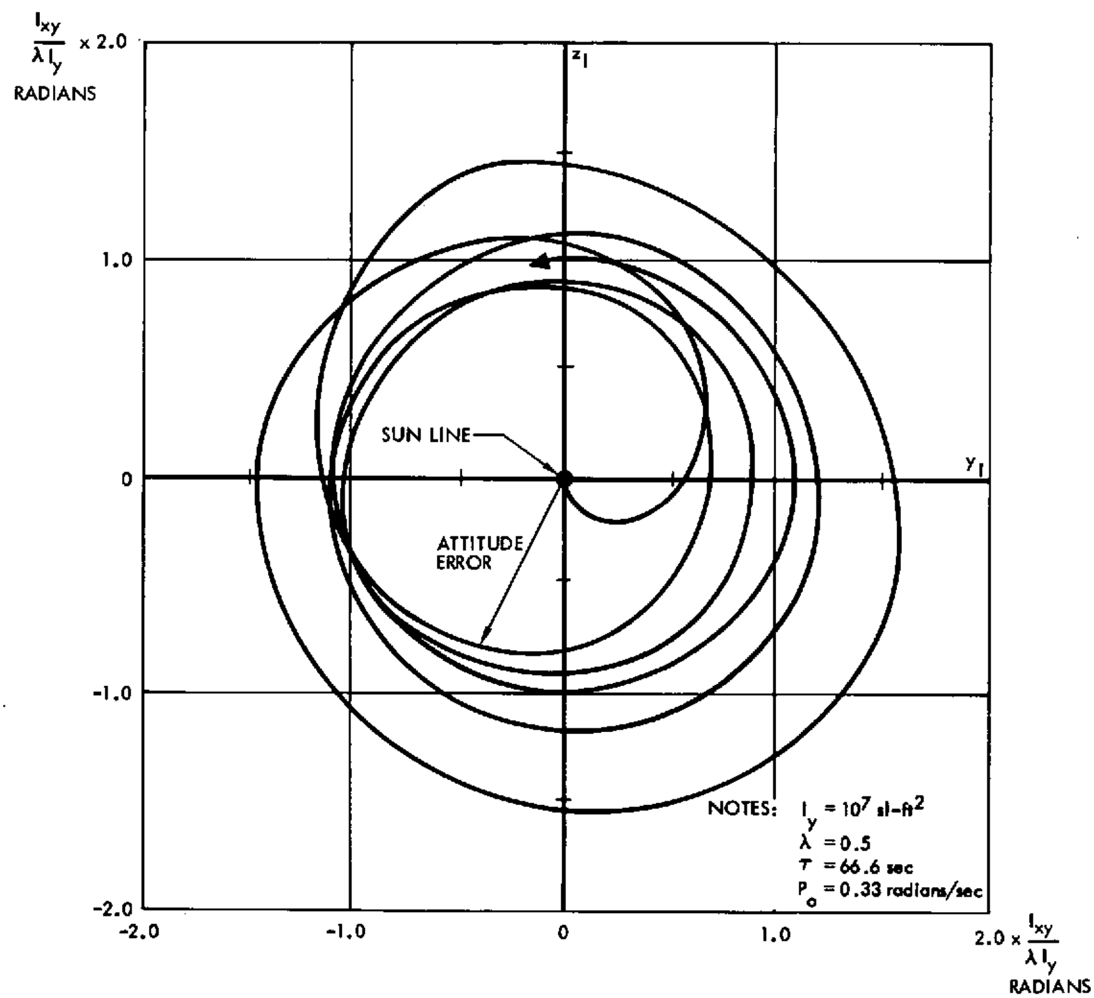


Figure 34. Inertial Motion of the Tip of a Unit Vector Along the Vehicle Spin Axis for Step-Product-of-Inertia Disturbance $[I_{xy}\ U(t)]$ - Rate Controlled System

$$z_I = -\frac{T_z}{I_y p_o}\left\{\frac{\cos(1+\lambda)\,p_o t}{(1+\lambda)} - \frac{1}{(1+\lambda)}\right\}$$

This is the equation of a circle of radius $T_z/I_y p_o(1+\lambda)$ and centered at 0, $T_z/I_y p_o(1+\lambda)$.

The undamped equations for a step torque $T_z\,U(t)$ (Figure 31) are

$$y_I = -\frac{T_z}{p_o^2 I_y \lambda}\left\{\frac{\cos(1+\lambda)\,p_o t}{(1+\lambda)} - \cos p_o t + \frac{\lambda}{(1+\lambda)}\right\}$$

$$z_I = -\frac{T_z}{p_o^2 \lambda I_y}\left\{\frac{\sin(1+\lambda)\,p_o t}{(1+\lambda)} - \sin p_o t\right\}$$

The equations of an epicycloid are of the same form as follows:

$$y_I = \left[(a+b)\cos\phi - a\cos\left(\frac{a+b}{a}\right)\phi\right]$$

$$z_I = \left[(a+b)\sin\phi - a\sin\left(\frac{a+b}{a}\right)\phi\right]$$

These are identical to the equations of motion if we set

$$a = -\frac{T_z}{p_o^2 I_y (1+\lambda)\lambda} = \text{radius of the moving generating circle}$$

$$b = -\frac{T_z}{p_o^2 I_y (1+\lambda)} = \text{radius of the fixed generating circle}$$

$$\phi = p_o t$$

and translate the vehicle epicycloidal path in the $-y_I$ direction by an amount equal to

$$\frac{T_z}{p_o^2 I_y (1+\lambda)}$$

In the case of a step change in product of inertia, a unit vector along the vehicle spin axis (Figure 32) would describe a path defined by the following expressions

$$y_I = \frac{I_{xy}}{\lambda I_y} \left\{ \cos p_o t - \cos (1 + \lambda) p_o t \right\}$$

$$z_I = \frac{I_{xy}}{\lambda I_y} \left\{ \sin p_o t - \sin (1 + \lambda) p_o t \right\}$$

DISTURBANCE ANALYSIS

The disturbances that will act on the vehicle may be classified as external or internal disturbances. The external disturbances are those that change the resultant angular momentum of the vehicle system, which includes the space station, its crew, and its internal systems. In general, the external disturbances must be countered through the application of an external torque on the vehicle by the attitude control system. Typical external disturbances are gravity gradient, docking impacts, and meteorite impacts. The internal disturbances are those that do not change the total system angular momentum but are capable of producing wobble (oscillatory accelerations throughout the vehicle). Typical internal disturbances are circulating fluids, rotating machinery, and dynamic mass unbalance. These disturbances can, in general, be countered by a mass conservative wobble damper such as the momentum wheel precession type.

Crew Motion

The disturbance effect on the vehicle caused by the motion of crew members is a problem best suited to computer analysis, primarily because of the time varying coefficients in the equations of motion. The results of such computer studies conducted at NASA - Langley show that crew motion can be a significant source of vehicle wobble. It is, therefore, important to consider the load imposed on the damper by such disturbances as a function of the frequency of occurrence. The prediction of crew motion frequency is an extremely formidable task; however, this prediction is only important if a non-mass conservative wobble damper, such as reaction-jet system, is used. The frequency with which crew motions take place will only affect the average power requirement of a momentum wheel precession system.

A second characteristic associated with crew movement is the variation in system mass distribution. Consider a case with the vehicle initially balanced and the spin axis aligned with the sun line. If the crew shifts in such a manner that they are not symmetrically distributed throughout the vehicle, then a product of inertia will exist. The vehicle motion, after the wobble has been damped, is characterized by the vehicle spinning about its new principal axis with the x body axis describing a cone about the sun line. The resulting attitude error has been plotted on Figure 35 as a function of product of inertia magnitude for several vehicle inertia combinations. The attitude error corresponding to a product of inertia for seven crew members standing together at the vehicle rim and 6 feet below the vehicle mass center is less than 1/2 degree.

Docking Impact

The disturbance effect of docking impacts can be readily determined if the disturbance is approximated by an impulse. The angular impulse imparted to the space station at impact is

$$T\delta(t) = mrv$$

Where

m = Mass of supply vehicle

r = Distance between the space station center of mass and the velocity vector of the supply vehicle measured normal to the velocity vector

v = Relative velocity between the vehicles

This angular impulse will change the total momentum of the system and result in an attitude error and a wobble. For reasonable closing velocities (1 or 2 fps) and for an Apollo-type supply vehicle with a mass of approximately 500 slugs, the propellant required to counteract the resulting attitude error is no more than 0. 2 lb per docking.

Gravity Gradient

The gravity gradient disturbance is a torque whose magnitude and direction are functions of the vehicle mass distribution and the orientation of the vehicle with respect to the radius vector from the center of the earth to the vehicle mass center. The torque exists unless the vehicle mass distribution is symmetrical about an axis along the aforementioned radius vector. As the spin axis of the space station will be directed toward the sun at all times, a gravity gradient torque will almost always exist. The following analysis was conducted to determine the approximate magnitude of this disturbance.

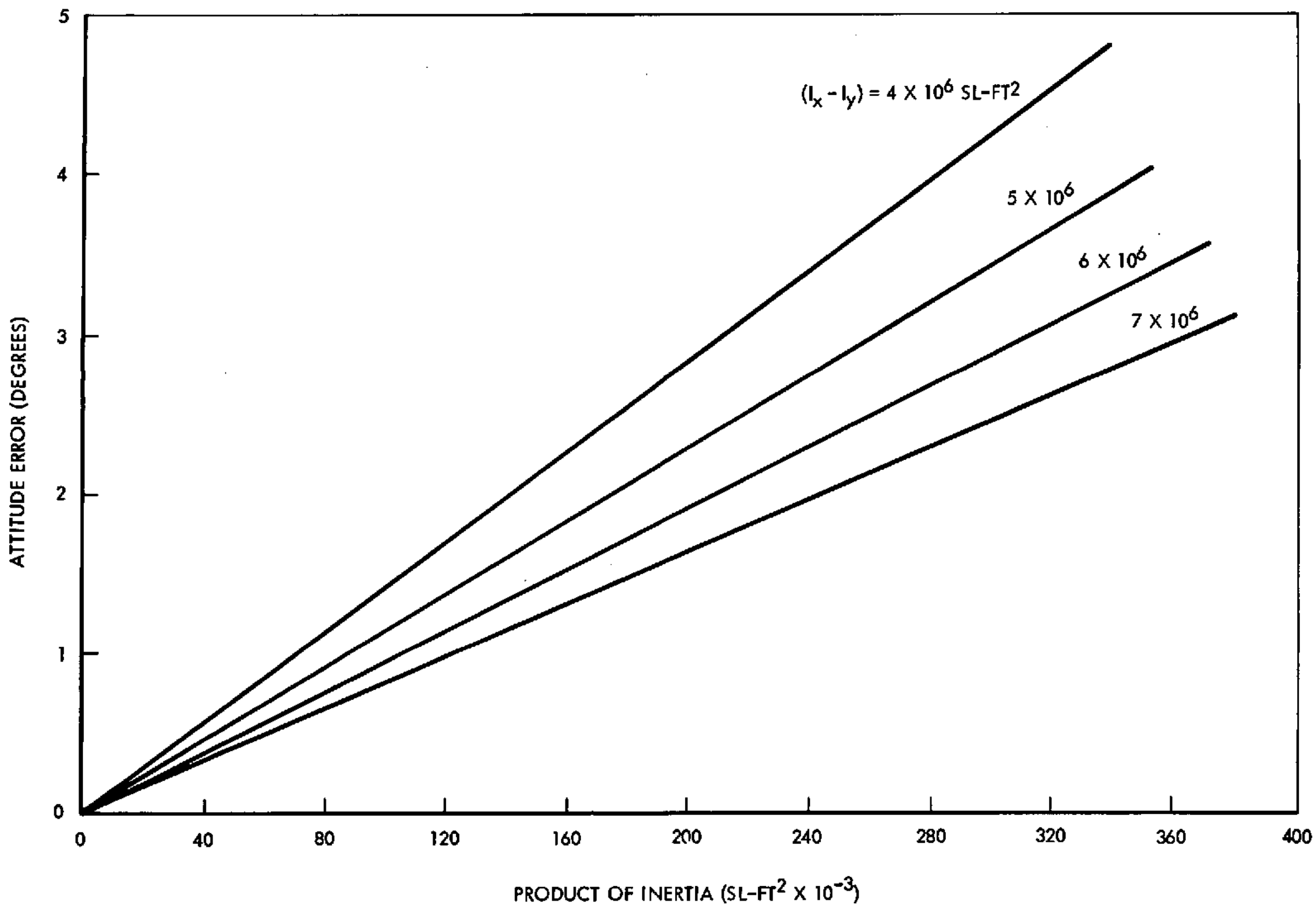


Figure 35. Attitude Error Associated With I_{xy} or I_{xz} Production of Inertia

Consider an orthogonal axis system x, y, z with the origin at the mass center of the vehicle and the x axis coincident with the x body or spin axis. Select the z axis to lie in the plane of the orbit. For the space station configuration, any axis that passes through the vehicle mass center and is also normal to the spin axis is a principal axis of inertia. As the z axis meets this criteria, it is a principal axis. The coordinate scheme is shown in the upper sketch on Figure 36. For this system, the torques exerted by gravity have been derived in Reference 1, and in our terminology the expressions are

$$T_{x_g} = 0$$

$$T_{y_g} = \frac{3\mu}{R^3}(I_x - I_y) \sin\alpha \sin u \cos u$$

$$T_{z_g} = \frac{3\mu}{R^3}(I_x - I_y) \sin^2 u \sin\alpha \cos\alpha$$

Where μ is the product of the earth's mass and the universal gravitational constant

α is the angle between the vehicle x body axis and the normal to the orbit plane

R is the distance between the center of the earth and the mass center of the vehicle

u is the angle between the orbital radius vector and the line of nodes

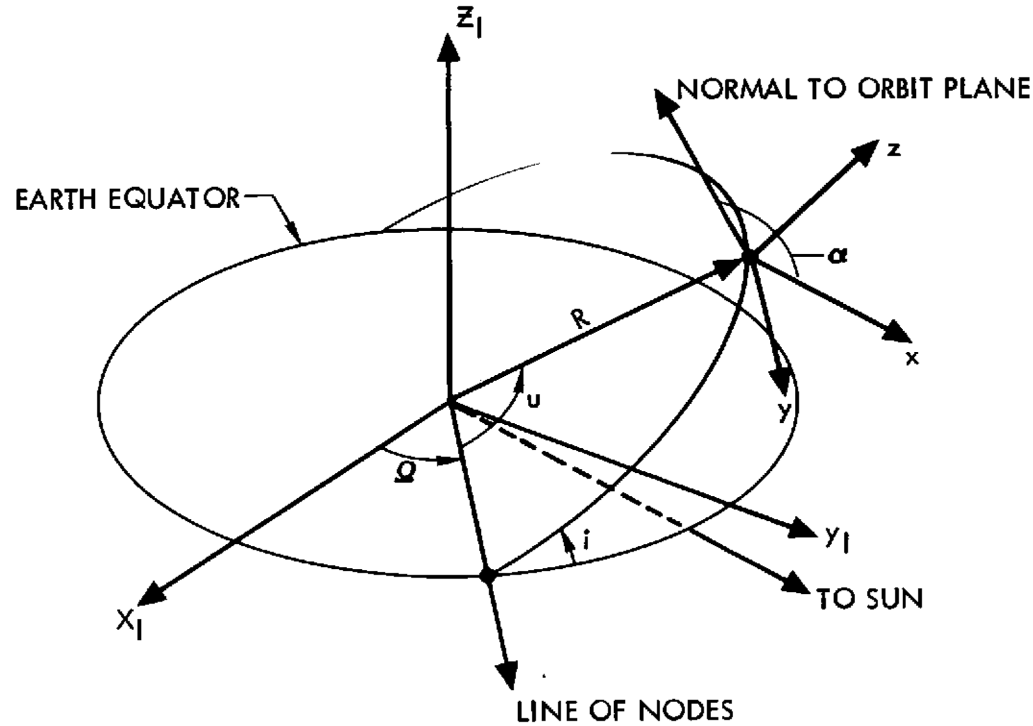
Through the use of trigonometric identities, the expressions for the torque about the y and z axes can be written in a more useful form.

$$T_{y_g} = \frac{3\mu}{2R^3}(I_x - I_y) \sin\alpha \sin 2u$$

$$T_{z_g} = \frac{3\mu}{4R^3}(I_x - I_y) \sin 2\alpha (1 - \cos 2u)$$

The maximum possible torque for T_{y_g} or T_{z_g} can be determined by inspection of the above equations and is seen to be

$$T_{g_{max}} = \frac{3}{2}\frac{\mu}{R^3}(I_x - I_y)$$



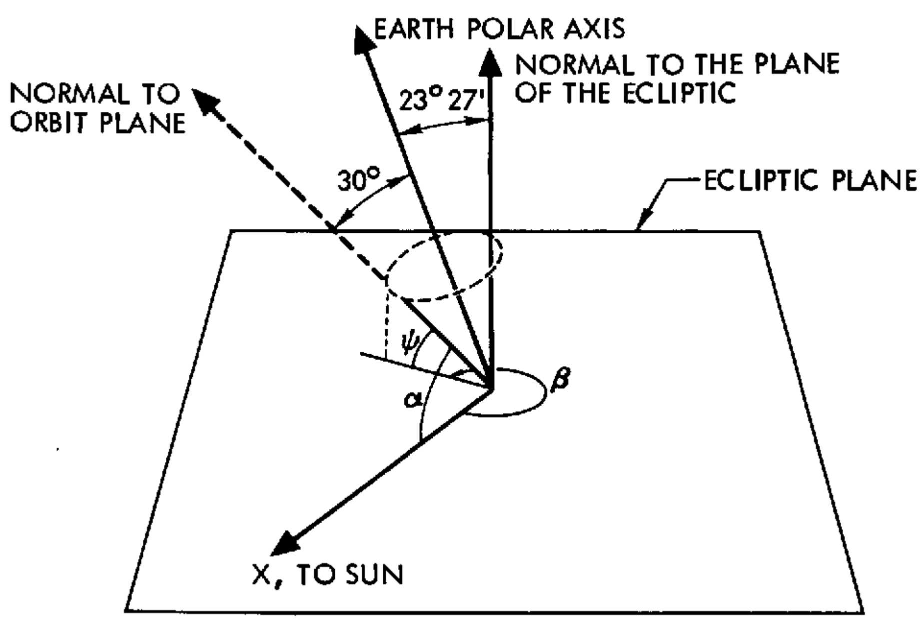


Figure 36. Geometry for Gravity Gradient Analysis

For a 300-nautical-mile orbit, the maximum torque is

$$T_{g_{max}} = \frac{3}{2} \frac{1.408 \times 10^{16}}{\left[22.75 \times 10^{6}\right]^{3}} (I_x - I_y)$$

$$= 1.79 \times 10^{-6} (I_x - I_y) \text{ ft-lb}$$

if

$$(I_x - I_y) = 5 \times 10^{6} \text{ sl-ft}^2$$

then

$$T_{g_{max}} = 8.95 \text{ ft-lb}$$

The calculated peak disturbance torque is large enough to warrant investigation of the torque magnitude as a function of time. Let us first consider the variation of T_{y_g} with time. As a first approximation $\dot{u} \equiv \omega$ where ω is the orbital rate. The torque about the y axis is a function of sin 2u and is therefore oscillating sinusoidally at approximately twice the orbital frequency. The average value of this component is zero. Let us next consider the torque about the z axis. This torque is composed of two terms, one of which is also a function of u and will therefore contribute no net torque. The remaining term in the T_{z_g} expression is a function of α, and this component will exert a net torque on the vehicle. To determine the magnitude of the component, it is necessary to know the variation of α with time. Consider the geometry shown on the lower sketch of Figure 36. Utilizing this geometry and an Euler transformation, the cosine of α may be approximated by the following expression.

$$\cos \alpha \cong \cos \left[\beta_o - (\omega_{sl} + \omega_{nodes}) t\right] \cos \left(60° + 23.45° \cos \omega_{nodes} t\right)$$

where

$$\omega_{sl} = 360°/\text{year} = 0.986°/\text{day}$$

$$\omega_{nodes} = 6.55°/\text{day (300-nautical-mile orbit with } 30^{\circ} \text{ inclination)}$$

A graphical technique is then used to estimate a conservative average value for the absolute magnitude of sin α and the result is

$$(\sin|2\alpha|)_{avg} = 0.54$$

The absolute magnitude is used because of the low frequency at which the term oscillates between positive and negative values. Utilizing the above results we can obtain a preliminary estimate of the weight of propellant required to counteract the gravity torque disturbance over a 1-year period.

$$W_p = \frac{3\mu(I_x - I_y)(0.54)(60)^2(24)(365)}{4R^3 \, l \, I_{sp}}$$

For

$$(I_x - I_y) = 7.6 \times 10^6 \text{ sl-ft}^2$$

$$l = 70 \text{ ft}$$

$$I_{sp} = 300 \text{ sec}$$

$$R = 22.75 \times 10^6 \text{ ft}$$

Then

$$W_p = 5520 \text{ lb}$$

It should be noted, in view of the assumptions made, that the above estimate is preliminary. It is apparent, however, that gravity torque is one of the major vehicle disturbances and a detailed analysis of the problem will be required.

Meteorite Impact

A preliminary analysis was made to determine the magnitude of attitude error that could result from the impact of meteorites on the vehicle. This investigation was based on the data presented in Reference 2 and included the effects of particles ranging from visual magnitude 10 to visual magnitude 30. The meteorites of magnitude smaller than 10 were excluded because of the low probability of impact. An extremely conservative estimate of the resulting attitude error was obtained by computing the expected number of impacts for each visual magnitude and the attitude deviation per impact. These deviations were then summed, and the resulting attitude error was found to be less than 6×10^{-4} degrees. It may, therefore, be concluded that meteorite impact does not represent a significant disturbance to the attitude control system.

Circulating Fluids and Rotating Machinery

Circulating fluids and rotating machinery can exert disturbing torques on any rotating vehicle. Specifically, rotating machinery in the space station will consist of such devices as pumps and fans for the environmental control system. Circulating fluids, used as heat transfer agents, will also be present in this system. The circulation of air in each of the modules also represents a possible source of disturbance torque.

An estimate of the maximum possible disturbance due to air circulation in one module was determined in the following manner.

With a module diameter of 10 feet and an effective module length of 63.5 feet, the volume of air per module is $\pi(25)(63.5) = 4990 \text{ ft}^3$. The density of air at 85 F and 14.7 psi is

$$\rho = 0.002378 \left(\frac{14.7}{14.7}\right)\left(\frac{519}{545}\right) = 0.00227 \text{ sl/ft}^3$$

The mass of air per module is then

$$4990\,(0.00227) = 11.3 \text{ slugs}$$

A preliminary estimate of the mass flow rate at the fan is

$$542 \text{ lb/hr or } 0.00468 \text{ sl/sec}$$

The air is assumed to be flowing in a closed rectangular path with dimensions of 10 feet by 63.5 feet with the flow rate constant along the path.

Total path length is

$$2\left[(63.5) + 10\right] = 147 \text{ ft}$$

Mass of air per ft is

$$\frac{11.3}{147} = 0.0769 \text{ sl/ft}$$

The velocity of flow is

$$0.00468 \frac{\text{slugs}}{\text{sec}} \times \frac{1}{0.0769} \frac{\text{ft}}{\text{slug}} = 0.0608 \text{ ft/sec}$$

The total angular momentum is then obtained by summing the momentum contribution of each path segment about the centroid of the assumed flow path.

Total $$H = \Sigma R_i m_i V_i$$

Where R_i is the distance from the centroid to the i^{th} path segment measured normal to the path, M_i is the mass of air in the i^{th} path segment, and V is the velocity of flow

Substituting in this equation,

$$H = 0.0608 \left[2(63.5)(0.0769) 5 + 2(10)(0.0769)(31.75)\right]$$

$$= 5.94 \frac{\text{sl/ft}^2}{\text{sec}}$$

If this vector is oriented normal to the vehicle spin axis and the vehicle is rotating at 0.339 rad/sec, the torque exerted on the vehicle is

$$T = \omega H = 0.339 (5.94) = 2.01 \text{ ft-lb}$$

This disturbance is significant and should be considered in the design layout of the environmental control system.

If the inertial direction of an angular momentum vector is to be changed, an external torque is required to accomplish this change. Thus, the angular momentum vectors of rotating machinery or circulating fluids can impose a disturbance on the vehicle if these vectors are forced to change direction in space as the vehicle rotates. However, if these momentum vectors lie parallel to the vehicle spin axis, their inertial direction is not affected by normal vehicle rotation. In other words, it appears desirable to mount the pumps and fans in such a way that the spin axis of their rotors is normal to the plane of the rim modules. A similar specification for circulating fluids would require that the path of circulation lie in a plane parallel to the plane of the rim modules.

A second method of minimizing the disturbance would entail opposing the momentum vectors of machines or fluids in one module with those in the diametrically opposite module. For example, consider the fans in two diametrically opposite modules. The fans would be mounted in such a way that the spin axes of their rotors are parallel, and the rotation of one rotor is opposite in sense to the rotation of the other.

For either of these proposed schemes, some disturbance torque will always be present due to inaccuracy in mounting the machinery and in deployment of the structure. This disturbance would easily be countered by a minute displacement of the precession wheel momentum vector.

Mass Unbalance

If the space station mass is distributed in such a way that products of inertia exist about the body axes, then in steady-state operation (all wobble damped), components of acceleration will exist normal to the desired artificial gravity. The magnitude of these undesired accelerations is a function of the product of inertia magnitude as shown in the analysis below.

Let us consider an I_{xy} product of inertia. The approximate expressions for the body rates that exist after all wobble has been damped are

$$p\ (t) = p_o$$

$$r\ (t) = o$$

$$q\ (t) = -\frac{p_o T_{xy}}{(I_x - I_y)}$$

The expressions defining the acceleration components existing at any point in the vehicle are

$$a_x = x(q^2 + r^2) + y\ (pq - \dot{r}) + z\ (pr + \dot{q})$$

$$a_y = x\ (pq + \dot{r}) - y\ (p^2 + r^2) + z\ (qr - \dot{p})$$

$$a_z = x\ (pr - \dot{q}) + y\ (qr + \dot{p}) - z\ (p^2 + q^2)$$

These acceleration components are measured in a coordinate system parallel to the body axis system and displaced from the mass center by the coordinates x, y, and z measured along the mass center body axes. The derivation of these equations may be found in Reference 3.

Substituting the body rates into the acceleration equations yields

$$a_x = -x \frac{p_o^2 I_{xy}^2}{(I_x - I_y)^2} - y \frac{p_o^2 I_{xy}}{(I_x - I_y)}$$

$$a_y = -x \frac{p_o^2 I_{xy}}{(I_x - I_y)} - y p_o^2$$

$$a_z = -z \left[p_o^2 + \frac{p_o^2 I_{xy}^2}{(I_x - I_y)^2} \right]$$

The second term in the expression for a_y and the first term in the espression a_z represent the desired artificial gravity. All other terms in the three expressions represent undesired components of acceleration. Considering the possible values for x, y, and z and reasonable magnitudes of $I_{xy}/(I_x - I_y)$, a suitable approximation for the maximum undesired acceleration component is $y_{max}\, p_o^2\, I_{xy}/(I_x - I_y)$. The desired acceleration is $y_{max}\, p_o^2$ and therefore the ratio of maximum undesired acceleration to desired acceleration is $I_{xy}/(I_x - I_y)$. As an example in the application of this criteria, assume that from a human factors standpoint accelerations no greater than 5 percent of the artificial gravity are tolerable in a direction normal to the design artificial gravity. From the analysis, I_{xy} must be less than 0.05 $(I_x - I_y)$. This criteria could be used to specify a tolerance on vehicle mass unbalance. Given a tolerable acceleration ratio and the minimum value of $(I_x - I_y)$ expected over the duration of the mission, a maximum I_{xy} is specified. The I_{xy} expected due to asymmetrical crew placement would then be subtracted from the maximum allowable and the remainder would specify the tolerable I_{xy} due to vehicle mass unbalance.

ATTITUDE REORIENTATION

A reorientation capability will be required to counter external disturbances on the vehicle and to maintain alignment with the solar reference. This capability can only be provided by an external source of torque. This technique appears quite promising and study is continuing in this area. In addition, a reaction-jet system was devised. It is based upon the use of constant thrust-level jets fueled with a storable hypergolic bipropellant and is described below.

System Description

There are several approaches that may be taken in the design of an attitude reorientation system. The system presented herein is a discontinuou type that applies an impulsive correction, allows the resulting wobble to be damped, and then determines if a further correction is required. A double dead band is used that will result in the following system operation. When the resultant attitude error exceeds the large dead band (a preset value, θdb_l, that is the maximum tolerable error) the system will continue to apply attitude corrections until the error has been reduced to a preset value of θdb_s. The object of using a double dead band is to prevent minor disturbances, such as small product-of-inertia variations from continuously activating the correction system. The logic used to integrate the reorientation system with the wobble damper is discussed in a separate section of this report.

Attitude Error Measurement

Sun sensors are used to measure the attitude error about the y and z body axes. For angles of the magnitude of interest, the sun sensor outputs may be considered linear. Let these outputs be represented by m and n, where m is proportional to the angular attitude error about the z body axis and n is proportional to the error about the y body axis. For small attitude errors about each of the body axes, the resultant attitude error θ, the angle between the x body axis and the sun line, is proportional to $\sqrt{m^2 + n^2}$.

Control Laws

When the system integration logic commands an attitude correction, the angular impulse applied to the vehicle by the jets is controlled in the following manner. The jets which torque the vehicle about the y body axis are commanded to remain on for a pulse duration proportional to m. In a like manner, an angular impulse proportional to n is applied about the z body axis.

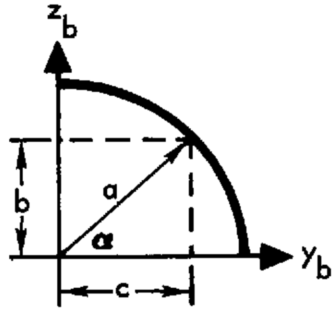
The resultant angular impulse, whose components are J_y and J_z, is in a direction such that the vehicle precesses to reduce the attitude error. The control laws are

$$J_y = Km$$
$$J_z = Kn$$

The system operation is shown in block diagram form on Figure 37.

Impulse Required For An Attitude Change

As the effective impulse applied to the vehicle is actually obtained by the vector sum of two components, the effective impulse is less than the arithmetic sum of the impulses applied by the jets. Consider the following sketch.



If the vector a represents the desired effective impulse, then b and c represent the impulse applied by the jets. If it is equally likely that the angle α will assume any angle between 0 and $\pi/2$, then we can compute an average value for (b+c) in terms of a.

$$(b+c) = a \sin \alpha + a \cos \alpha$$

$$(b+c)_{avg} = \frac{a}{\frac{\pi}{2}} \int_0^{\pi/2} \sin \alpha d\alpha + \int_0^{\pi/2} \cos \alpha d\alpha = \frac{4}{\pi} a$$

The impulse which must be supplied by the reaction jets is therefore, on the average, $4/\pi$ times as much as the effective impulse required.

The effective impulse required, J_e can be readily determined as follows.

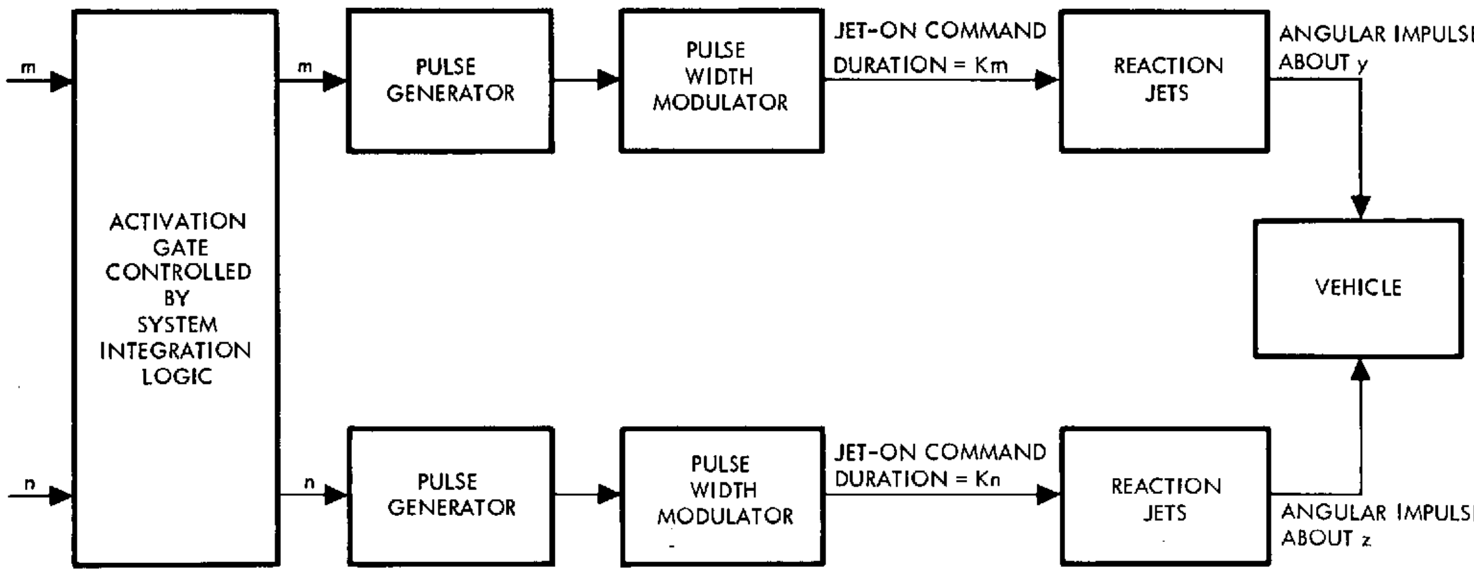


Figure 37. System Diagram of Attitude Reorientation Scheme

$$T = \omega \times H$$

$$\Delta\theta = \omega t$$

$$J_e = Tt$$

where

T = Applied torque

ω = Precession rate

H = Vehicle angular momentum

$\Delta\theta$ = Attitude change

Combining the above equations

$$J_e = \Delta\theta (H)$$

The impulse supplied by the jets, J_j, is then

$$J_j = \frac{4}{\pi} (\Delta\theta) H$$

And the weight of propellant required is

$$W_p = \frac{4 (\Delta\theta) H}{\pi l I_{sp}}$$

where

l = Jet lever arm

I_{sp} = Propellant specific impulse

Using this expression, an estimate may be obtained for the weight of propellant required to track the sun over a period of 1 year. Using the following parameter magnitudes

$$\Delta\theta = 2\pi$$

$$p_o = 0.34 \text{ rad/sec}$$

$$I_x = 15 \times 10^6 \text{ sl-ft}^2$$

$$l = 70 \text{ ft}$$

$$I_{sp} = 300 \text{ sec}$$

the weight of propellant required is 1945 pounds. The variation in this propellant requirement as a function of artificial gravity level is shown on Figure 38.

A summary showing the estimated attitude control system propellant required to counter the various external disturbances for two different mission durations is shown in Table 9. For mission durations longer than the periods stated, propellant resupply would be required.

Table 9. Propellant Requirement Summary for Reorientation and Spin Control

Task	Weight of Propellant Required	
	12-Week Mission	1-Year Mission
Spin-up	250	250
Spin Rate Experimentation	330	330
Track Sun	500	2150
Gravity Gradient	1270	5520
Docking Impacts	-	20
	2350	8270

Pulse Duration and Jet Size Considerations

In order to maintain a reasonable accuracy of the direction of the resultant precession, the jet pulse duration must be kept small. If the jet on-time is limited to the corresponding time for 10 degrees of vehicle rotation, which should result in reasonable accuracy, then the maximum pulse duration for a vehicle rotating at 0.34 rad/sec is approximately 0.514 seconds. The only known continuous external disturbance acting on the vehicle which can cause an inertial attitude drift is gravity gradient. Therefore, the jets should be sized so that the system is at least capable of tracking this disturbance. The peak gravity-gradient torque, as computed in the disturbance section of this report, is about 9 foot-pounds. The precession rate corresponding to this torque is approximately 1.0×10^{-4} deg/sec. If it is conservatively assumed that the wobble damper requires 3 minutes to

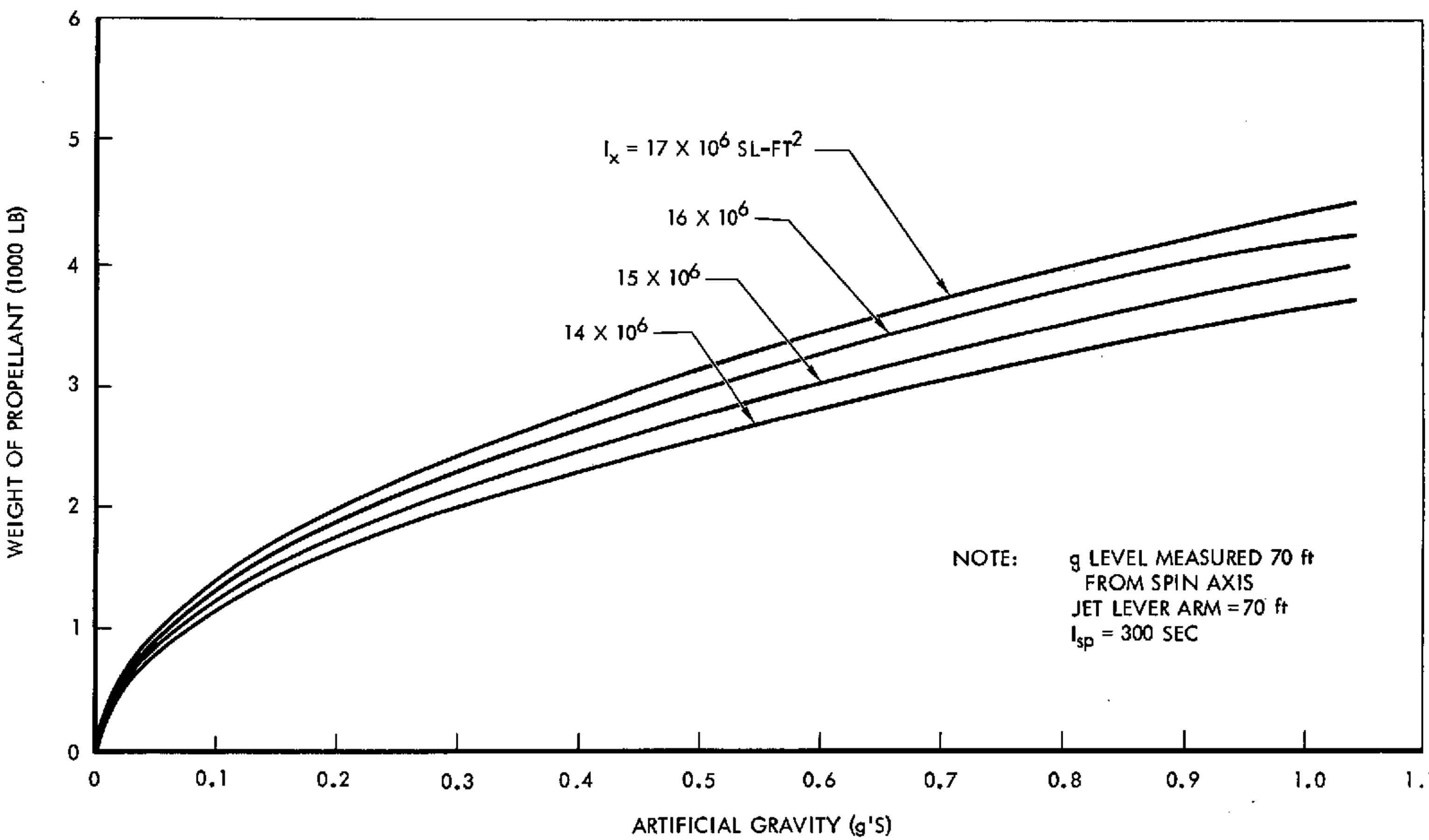


Figure 38. Weight of Reaction Jet Propellant Required to Track the Sun

damp the wobble associated with a corrective impulse, this time is then the minimum time between corrections. The attitude drift over this time period would be 0.018 degrees. Allowing a safety factor of about 2, the jets should then be capable of changing the attitude by 0.04 degrees per pulse. If the jets are torquing as a couple, with a lever arm of 70 feet on each jet, and the pulse duration as previously specified is 0.514 seconds, the required thrust level is approximately 50 pounds.

WOBBLE DAMPING

A mass conservative precession-wheel damper was studied; and estimates of system weight, power, and torque requirements were obtained. A reaction-jet damper was also analyzed. System gain and thrust level requirements were estimated, and an expression was developed for the weight of propellant required to damp an impulsive disturbance.

Control Law Philosophy

The development of the proper control laws is based on the following objectives. Regardless of the type system used to control attitude and damp wobble, the vehicle steady-state condition should be one of stable equilibrium; in addition, after the damping, its momentum vector should be aligned with the system momentum which existed just following application of an impulsive disturbance. The first condition is mandatory with the reaction jets, since they would otherwise continue to be operative (for product-of-inertia disturbance) even after the transient motion had been damped. Obviously, the fuel expenditure would be intolerable. If the jets were turned off after damping, a residual wobble would remain. In the case of the precession wheel wobble damper, unstable equilibrium would require a continuous torque and, hence, a constant power drain from the electrical power supply (same type disturbance). Clearly, this is undesirable.

Consider a mass unbalance applied stepwise to the vehicle, and use the following control laws for the jet and wheel systems respectively:

$$T_{yc} = -K_1 q$$

$$T_{zc} = -K_1 r$$

and

$$\gamma_y = K_1 q$$

$$\gamma_z = K_1 r$$

The steady-state motion of the vehicle spin axis describes a cone about the total angular momentum vector, and the angle included by this cone is somewhat smaller than that dictated by the original shift in the principal axis. This is the unstable equilibrium condition formerly mentioned because the resultant spin vector, though aligned with the total momentum vector (no wobble), is not along the new principal axis of the vehicle.

In an effort to overcome this shortcoming, the following control laws were assumed for the jet system:

$$T_{yc} = -K_3 \dot{r}$$

$$T_{zc} = K_3 \dot{q}$$

Considering the precession-wheel system, these relations correspond exactly to

$$\gamma_y = \left(\frac{K_3}{H}\right) q - \left(\frac{K_3 p_o}{H}\right) m - n$$

$$\gamma_z = \left(\frac{K_3}{H}\right) r - \left(\frac{K_3 p_o}{H}\right) n + m$$

Now, regardless of the type disturbance, the vehicle wobble is damped out and no control torque is required to maintain the steady-state condition. In the case of an impulsive-torque disturbance, the total angular momentum of the system abruptly assumes a new magnitude and direction at t = 0+. After damping, the vehicle momentum vector is not quite aligned with this new angular momentum vector. As a matter of fact, the higher the value of K_3, the closer does the final attitude of the vehicle momentum vector approach that of the momentum vector that existed prior to the disturbance. Inasmuch as fuel expenditure computed per unit torque impulse is of the same order of magnitude for both the acceleration controlled and the rate controlled systems, a reaction-jet damper operating on acceleration signals would show definite advantage. However, this might not be true for the precession-wheel damper. The precession wheel cannot shift the overall system momentum vector in inertial space, because it cannot apply any external forces to the system. Thus, the system's resultant angular momentum is at all times constrained to lie along a line determined by the vector sum of the original vehicle momentum and that added by the disturbance impulse. This means that if the steady-state attitude of the vehicle momentum deviates from this line, the precession wheel momentum cannot be colinear with the vehicle spin axis. As will be shown later, this condition causes the precession-wheel gimbal angles to oscillate at the vehicle spin frequency. This is not satisfactory since the allowable misalignment of the spin axes is definitely limited. In addition, this is not conducive to long gimbal bearing life. Further study will be conducted in an effort to determine whether modifications to the acceleration laws are required. Ideally, the control laws should be formulated so as to damp body accelerations to zero and the gimbal angles to constant values.

Precession-Wheel Wobble Damper

System Description

Two or more precession wheels are suggested in order to provide sufficient system reliability. For example, with a two-wheel system, the wobble damping time constant would be doubled if one unit became inoperative; with three wheels the time constant would be increased by only 50 percent. Therefore, the subsequent torque and power determinations made for a single-unit system should be considered only as typical.

Each wheel is mounted on a spherical center bearing and the effect of gimbaling is obtained by the use of four hydraulically actuated plungers. These are spaced at 90-degree intervals around the wheel and bear against the center nonrotating portion. The actuators are placed so that one pair is able to move the wheel through δ_y radians about the y body axis, and the other pair provides similar action through δ_z radians about the z body axis. This is shown on the configuration diagram.

During wheel spin-up, all four actuators will be extended by equal amounts so that the wheel spin axis is aligned with that of the vehicle. It is estimated that approximately 150 watts per wheel will be needed to maintain wheel spin.

Vehicle body rates will be measured by rate gyros. The instrumentation axes are aligned with the vehicle's y and z axes. In the case of the acceleration control laws, the sun sensor is aligned so that its two outputs, m and n, are proportional to the vehicle angular attitude error about the vehicle y and z axes respectively.

Wheel Size Determination

If this system is to be used, a fairly large value of angular momentum is needed in the wheel(s), and the weight must not be excessive. If the angle subtended by the wheel rim is θ_w radians, then the total gimbal freedom is reduced by this amount. Assuming a trapezoidal cross section for the wheel, the following expression for its weight as a function of the radii, θ_w, and the density can be obtained:

$$W = 8.37\, \theta_w \left\{ D_2^{\ 3} - D_1^{\ 3} \right\} \rho \quad \text{lb}$$

where D_1 and D_2 are the inner and outer radii. The maximum allowable speed in terms of the allowable stress, the weight, density, and radii is

$$N = \frac{(S/\rho)^{1/2} \times 0.38 \times 10^2}{\left\{ 3.3 D_2^{\ 2} + .7 D_1^{\ 2} \right\}^{1/2}} \quad \text{rpm}$$

Finally, the wheel angular momentum as a function of the weight, speed, and radii is

$$H = 41.2 \times 10^{-4} N \theta_w \rho \left\{ D_2^{\ 5} - D_1^{\ 5} \right\} \quad \text{sl-ft}^2/\text{sec}$$

A typical momentum (Figures 39 and 40) for one steel wheel might be 20,000 sl-ft^2/sec, where D_1 = 4 feet 11 inches; D_2 = 5 feet; W = 200 pounds and N = 4500 rpm. This rotational speed corresponds to a 6 percent slip for a 10-pole 400~ induction motor and so would permit direct coupling of the spin motor to the wheel without the necessity of gearing. This arrangement is shown on the configuration diagram. The foregoing dimensions will be used in subsequent examples of peak gimbal swing, torque, and power requirements.

General Equations

As with the vehicle, the reactive torques generated by the precession wheel(s) can be written as a function of the angular momentums. Thus,

$$T_{ci} = - \left. \frac{d\overline{H}}{dt} \right|_i = - \frac{\partial H_i}{\partial t} - (\omega \times H)_i$$

The x, y, and z components of the control wheel angular momentum vector have rates as follows:

$$\omega_x = p_o$$

$$\omega_y = q$$

$$\omega_z = r$$

where the corresponding components of the wheel angular momentum are

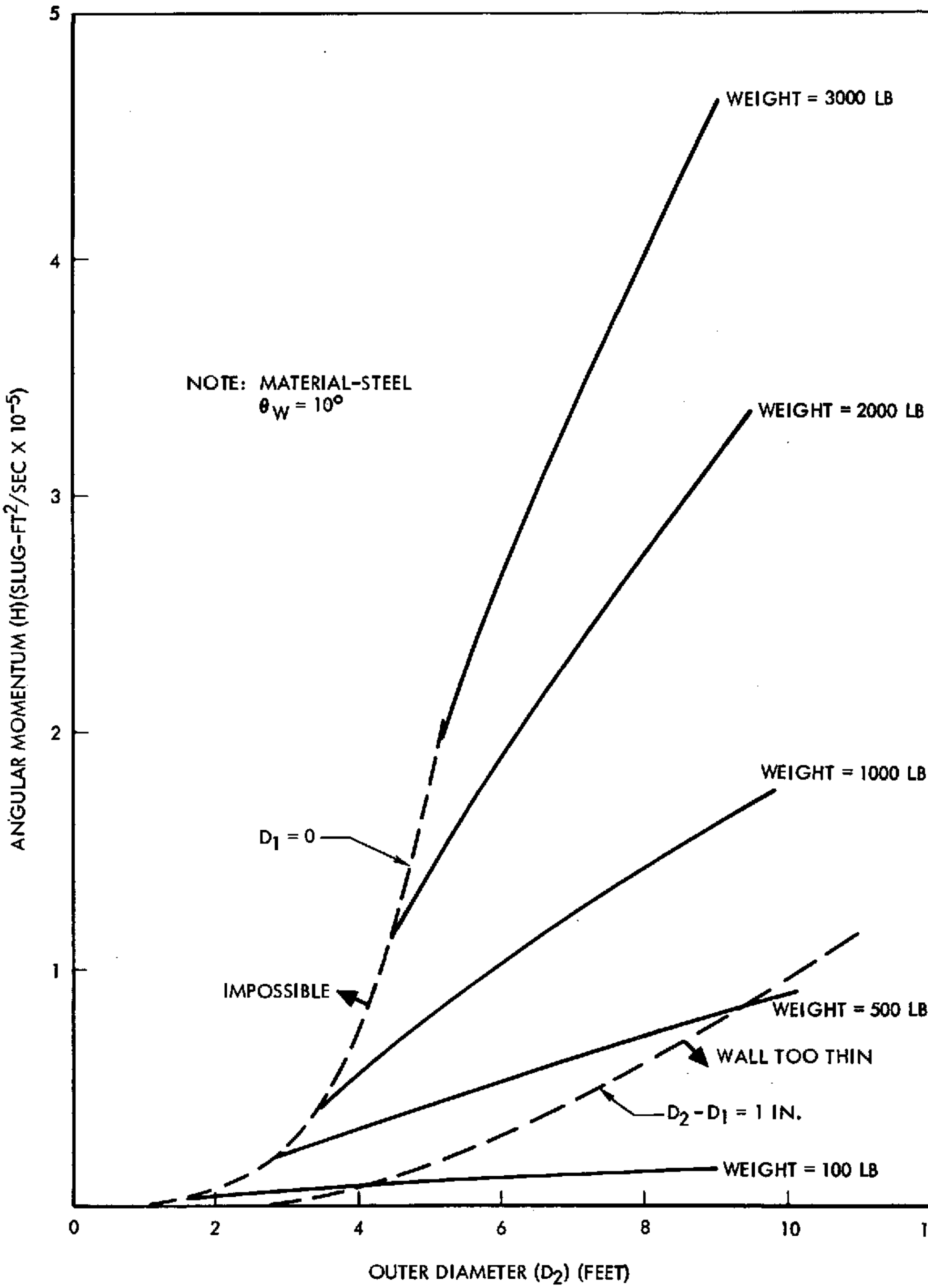


Figure 39. Precession Wheel Angular Momentum Versus Outer Diameter and Weight

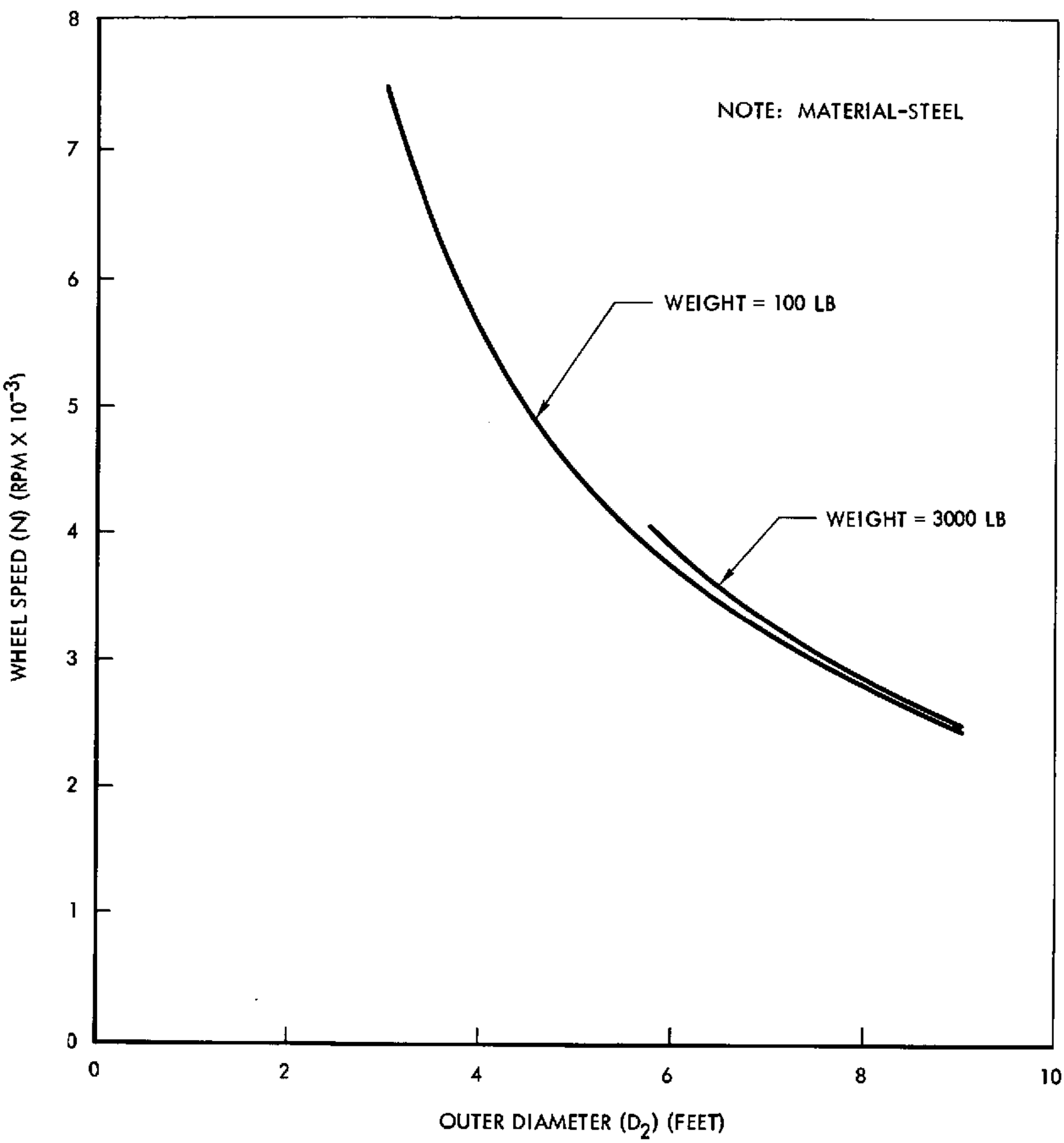


Figure 40. Precession Wheel Speed Versus Outer Diameter and Weight

$$H_x = H\left[1 + \tan\gamma_y + \tan^2\gamma_z\right]^{-1/2} \cong H$$

$$H_y = H(\tan\gamma_z)\left[1 + \tan^2\gamma_y + \tan^2\gamma_z\right]^{-1/2} \cong H\gamma_z$$

$$H_z = -H(\tan\gamma_y)\left[1 + \tan^2\gamma_y + \tan^2\gamma_z\right]^{1/2} \cong -H\gamma_y$$

Performing the vector differentiation indicated above yields the following torque equations:

$$T_{xc} = H\gamma_y q + H\dot{\gamma}_y\gamma_y + H\gamma_z r + H\dot{\gamma}_z\gamma_z \approx 0$$

$$T_{yc} = -H\dot{\gamma}_z - Hp_o\gamma_y - Hr$$

$$T_{zc} = H\dot{\gamma}_y + Hq - H\gamma_z p_o$$

This development assumes small gimbal angles. As will be shown later, the peak gimbal swings will not exceed 20 degrees. Thus, the assumption that $\tan\gamma \cong \gamma$ is in error by approximately 4 percent. The assumption that $[1 + \tan^2\gamma_y + \tan^2\gamma_z] \cong 1$ causes a simultaneous error in all the generated torques. This discrepancy is never more than 12 percent and is in such direction as to cause a small percentage increase in the wobble damping time constant.

The first of these equations cannnot be La Place transformed because of its highly nonlinear character. However, under the assumption of small perturbations, $T_x \cong 0$. The other two can be transformed as follows:

$$T_{yc} \cong -Hs\,\gamma_z - Hp_o\gamma_y - Hr$$

$$T_{zc} \cong Hs\,\gamma_y + Hq - Hp_o\gamma_z$$

Rate-Controlled System

A system diagram and a block diagram of the rate-control system are given in Figures 41 and 42 respectively. If the gimbal control laws are assumed to be:

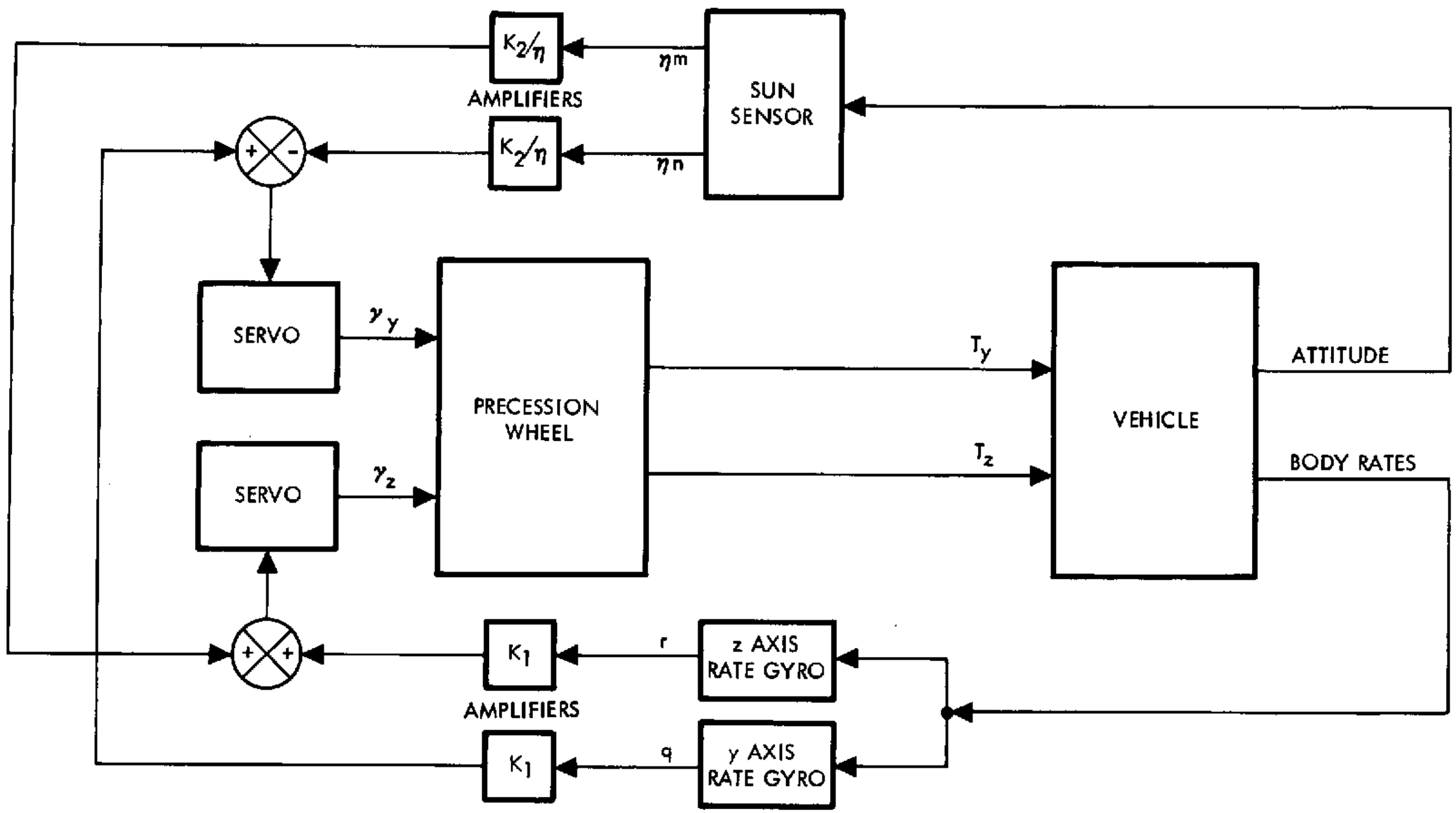


Figure 41. System Diagram - Rate-Controlled Precession Wheel Wobble Damper

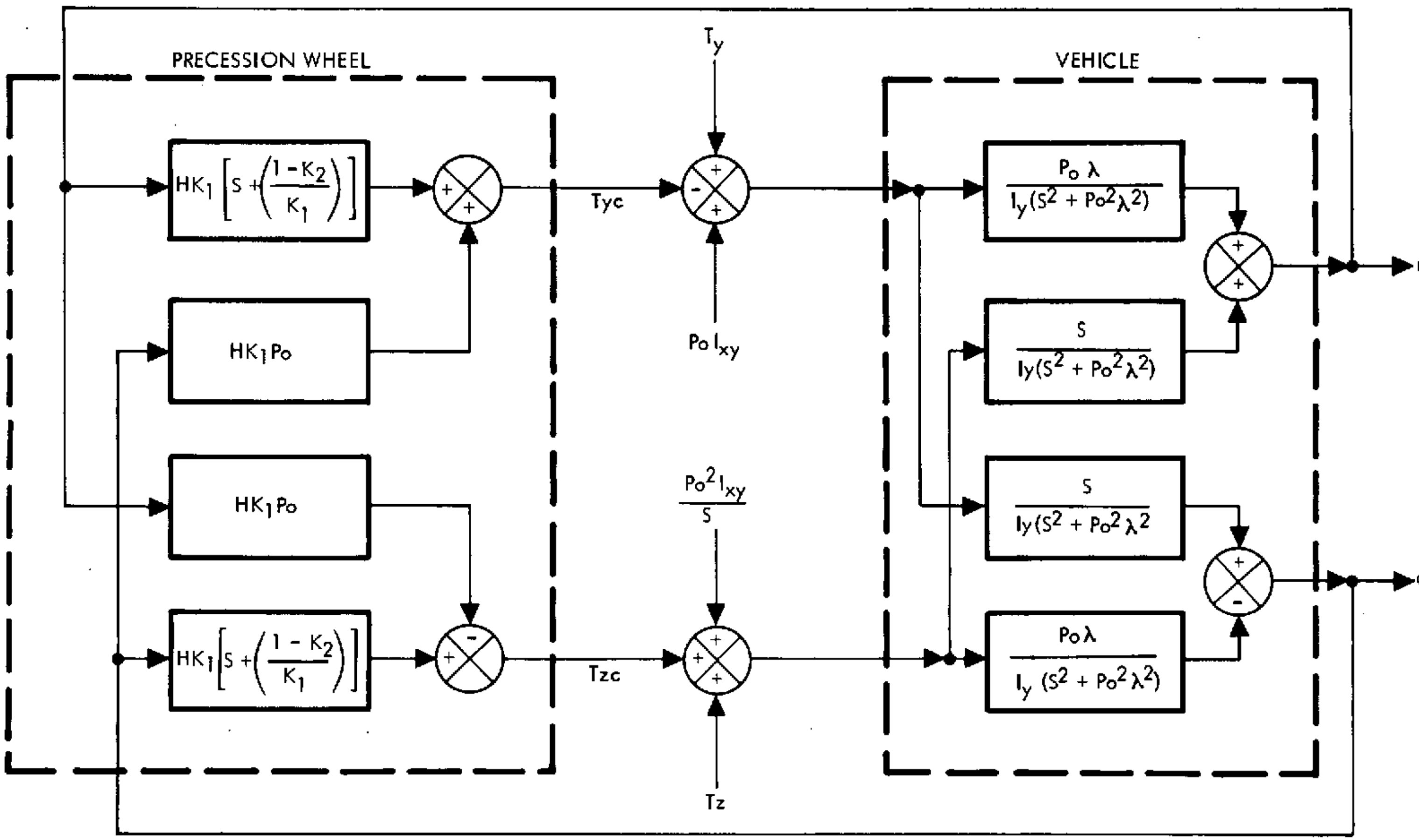


Figure 42. Block Diagram - Rate-Controlled Precession Wheel Wobble Damper

$$\gamma_y = K_1 q - K_2 n$$

$$\gamma_z = K_1 r + K_2 m$$

then the torques which are developed by the wheel are

$$T_{yc} = -HK_1 \left\{ p_o q + \left(s + \frac{1 - K_2}{K_1} \right) r \right\}$$

$$T_{zc} = HK_1 \left\{ \left(s + \frac{1 - K_2}{K_1} \right) q - p_o r \right\}$$

Consider an impulsive torque T_y applied to the vehicle. Then the body rates are

$$q = \frac{T_y I_y \left\{ s + \frac{p_o HK_1}{I_y} \right\}}{\left\{ I_y^2 + (HK_1)^2 \right\} \left\{ s^2 + \frac{2HK_1 \left[(1+\lambda) I_y p_o + H(1-K_2) \right]}{I_y^2 + (HK_1)^2} s + \frac{\left[p_o \lambda I_y + H(1-K_2) \right]^2 + (HK_1 p_o)^2}{I_y^2 + (HK_1)^2} \right\}}$$

$$r = \frac{T_y HK_1 \left\{ s + \frac{p_o \lambda I_y + H(1 - K_2)}{HK_1} \right\}}{\left\{ I_y^2 + (HK_1)^2 \right\} \left\{ \text{characteristic equation} \right\}}$$

By applying the final value theorem in the La Place domain we find that both q and r go to zero in the steady-state regardless of what control system parameters values are chosen, which indicates that the wobble is damped.

From the characteristic equation, the system time constant for the particular configuration under study is found to be

$$\tau = \frac{I_y{}^2 + (HK_1)^2}{HK_1\left[(1+\lambda) I_y p_o + H(1 - K_2)\right]}$$

$$= \frac{10^{14} + (HK_1)^2}{HK_1\left[1.5 \times 10^7 \times 1/3 \times H(1 - K_2)\right]} \text{ sec}$$

Assuming $HK_1 \ll 10^{14}$ and $H(1 - K_2) \ll 5 \times 10^6$ we obtain

$$\tau \cong \frac{20 \times 10^6}{HK_1} \text{ sec}$$

Assuming further that $H(1 - K_2) \ll p_o \lambda I_y$, then the system natural frequency is

$$\omega_n \cong p_o \lambda \text{ rad/sec}$$

The attitude deviation as measured along the y body axis is

$$m = \frac{-T_y HK_1 \left\{s^2 + \left[\frac{p_o(\lambda - 1)I_y - H(1 - K_2)}{HK_1}\right]s - p_o{}^2\right\}}{\left\{I_y{}^2 + (HK_1)^2\right\}\left\{s^2 + p_o{}^2\right\}\left\{\text{characteristic equation}\right\}}$$

The inverse transform of this equation reveals an interesting point:

$$m(t) = \frac{-T_y}{\left\{p_o I_x + H(1 - K_2)\right\}} \sin(p_o t + \psi) + \text{transient term}$$

The magnitude of the steady-state sinusoidal term is approximately the angle deviation between the vehicle and the sun line. However, at t = 0+ the impulsive torque causes the total system momentum vector to shift by

$$\frac{T_y}{p_o I_x + H}$$

Inasmuch as the precession wheel damper is unable to move this vector in any way, the vectorial sum of the wheel momentum and the vehicle momentum must always lie in this direction. Thus, because of the K_2 factor, a residual angle remains between the wheel and vehicle spin axes, a condition which can only be satisfied by the wheel oscillating about the gimbals at the vehicle spin frequency. This is an undesirable situation. Therefore, it is suggested that the attitude signals not be used in the manner shown.

Now consider the application of a step of torque $T_y U(t)$; the vehicle body rates are given by

$$q = \frac{T_y I_y \left\{ s + \frac{p_o H K_1}{I_y} \right\}}{\left\{ I_y^2 + (HK_1)^2 \right\} s \left\{ \text{characteristic equation} \right\}}$$

$$r = \frac{T_y H K_1 \left\{ s + \frac{p_o \lambda I_y + H}{HK_1} \right\}}{\left\{ I_y^2 + (HK_1)^2 \right\} s \left\{ \text{characteristic equation} \right\}}$$

The corresponding time response would include a constant term and a damped oscillation.

The steady-state values of r and q are constant (which indicates that the vehicle wobble is damped) and are equal to

$$q_{ss} = \frac{T_y p_o H K_1}{(I_y p_o \lambda + H)^2 + (HK_1 p_o)^2}$$

$$r_{ss} = \frac{T_y \left\{ p_o \lambda I_y + H \right\}}{(I_y p_o \lambda + H)^2 + (HK_1 p_o)^2}$$

As can be seen, the effect of the control system is to decrease these rates. However, the decrease is at the expense of a continuous power and torque requirement.

With a step of product of inertia, $I_{xy}U(t)$, applied to the vehicle, the vehicle body rates are given by

$$q = \frac{I_{xy} p_o I_y \left\{ s^2 - p_o \left[p_o \lambda + \frac{H}{I_y} \right] \right\}}{\left\{ I_y^2 + (HK_1)^2 \right\} s \left\{ \text{characteristic equation} \right\}}$$

$$r = \frac{I_{xy} HK_1 p_o \left\{ s^2 + \left[\frac{p_o (1 + \lambda) I_y + H}{HK_1} \right] s + p_o^2 \right\}}{\left\{ I_y^2 + (HK_1)^2 \right\} s \left\{ \text{characteristic equation} \right\}}$$

The time response of these signals contains a constant term and a damped sinusoid. Again, the steady-state constant term indicates stable wobble damping.

The equations of the vehicle attitude referred to the body axes are

$$m = \frac{-I_{xy} HK_1 p_o \left\{ s + \frac{p_o \lambda I_y + H}{HK_1} \right\}}{\left\{ I_y^2 + (HK_1)^2 \right\} s \left\{ \text{characteristic equation} \right\}}$$

$$n = \frac{-I_{xy} p_o I_y \left\{ s + \frac{p_o HK_1}{I_y} \right\}}{\left\{ I_y^2 + (HK_1)^2 \right\} s \left\{ \text{characteristic equation} \right\}}$$

The corresponding time equations would again indicate a constant and a damped sinusoid. The final values of these equations are

$$m_{ss} = \frac{-I_{xy} p_o \left[p_o I_y \lambda + H \right]}{\left[p_o \lambda I_y + H \right]^2 + \left[HK_1 p_o \right]^2}$$

$$n_{ss} = \frac{-I_{xy} p_o \left[p_o HK_1 \right]}{\left[p_o \lambda I_y + H \right]^2 + \left[HK_1 p_o \right]^2}$$

The steady-state angle between the vehicle spin axis and the sun line is given by

$$\theta = \sqrt{m^2_{ss} + n^2_{ss}}$$

$$\theta = \frac{I_{xy} p_o}{\left\{ \left[p_o \lambda I_y + H \right]^2 + \left[H K_1 p_o \right]^2 \right\}^{\frac{1}{2}}}$$

Note that this angle is decreased by the H terms but at the expense of constant torque and power requirements. No setting can be found for K_1, which would eliminate this problem.

Computer studies at NASA Langley Research Center have indicated that a continuous tangential crew motion results in an attitude error that is somewhat amplified over that obtained by a step change in product of inertia. This magnification was a function of the ratio of crew walking rate to vehicle spin rate. When this ratio approached λ, then large magnification factors were involved. The magnification was found to be less for walking rates smaller than or greater than this resonant rate. This amplification is being studied in detail so that the proper degree of safety factor can be designed into the control system for any given walking rate.

If a system time constant of 66 seconds (approximately three and one-half spin periods) is assumed, for H = 20,000 and K_1 = 15, the gimbal angle variation for a step product of inertia is

$$\delta_y = K_1 q$$

$$= I_{xy} \times 10^{-6} \left\{ -1.0 + (1 + \lambda) \left(\exp. \left[- \frac{HK_1 \times 10^{-6} (1 + \lambda) p_o t}{I_y} \right] \right) \cos p_o \lambda t \right\}$$

$$= I_{xy} \times 10^{-6} \left\{ -1 + 1.5 e^{-0.015t} \cos 0.168t \right\} \text{ rad}$$

The peak gimbal angle occurs at t ≅ 10 sec and its magnitude is

$$\gamma_y \max = -1.08 \, I_{xy} \times 10^{-6} \text{ radians}$$

To the extent that the inequalities indicated regarding H are valid, the peak gimbal angle is relatively independent of H. The gimbal angle rate is

$$\dot{\gamma}_y = I_{xy} \times 10^{-6} \times 1.47e^{-0.015t} \left\{ -0.015 \cos 168\,t - 0.168 \sin 168\,t \right\} \text{rad/sec.}$$

The torque required at the y gimbal is essentially equal to T_{yc} and was given earlier as

$$T_{yc} = (HK_1) \left\{ p_o q + \left(s + \frac{1}{K_1} \right) r \right\}$$

$$= 300\, I_{xy} \times 10^{-6} \left\{ \delta(t) - 19.28 + 47.9e^{-0.015t} \cos 0.167t \right\} \text{lb.-ft.}$$

Neglecting the impulse term, it is found that the maximum torque occurs at t = 0+ and is equal to $8400 \times 10^{-6}\, I_{xy}$ ft-lb. Thus, the actuator through any gearing must be capable of this stall torque. The impulse term at t = 0+ arises from the abruptness of the product of inertia disturbance. In reality, product-of-inertia disturbances (i.e. crew motion) will have very slow variations. Therefore, this impulse can be ignored. Note that a constant torque requirement of $5760\, I_{xy} \times 10^{-6}$ remains after the transient has been damped. A more judicious choice of control laws might eliminate this without causing other difficulties.

The power needed to counter the product-of-inertia disturbance can be computed as a product of the gimbal torque and the gimbal rate.

$$P = -I_{xy}^{2} \times \frac{746}{550} \times 1.47 \times 10^{-6} \times 300e^{-0.015t} \left\{ 0.015 \cos 0.167t \right.$$

$$\left. +0.167 \sin 0.167t \right\} \left\{ -19.28 + 47.9e^{-0.015t} \cos 0.167t \right\} \text{watts}$$

The maximum value of this expression occurs at t = 13 sec and is approximately $270 \times 10^{-12}\, I_{xy}^{2}$ watts.

Acceleration-Controlled System

A system diagram and a block diagram of the acceleration-controlled wobble damper are given in Figures 43 and 44 respectively.

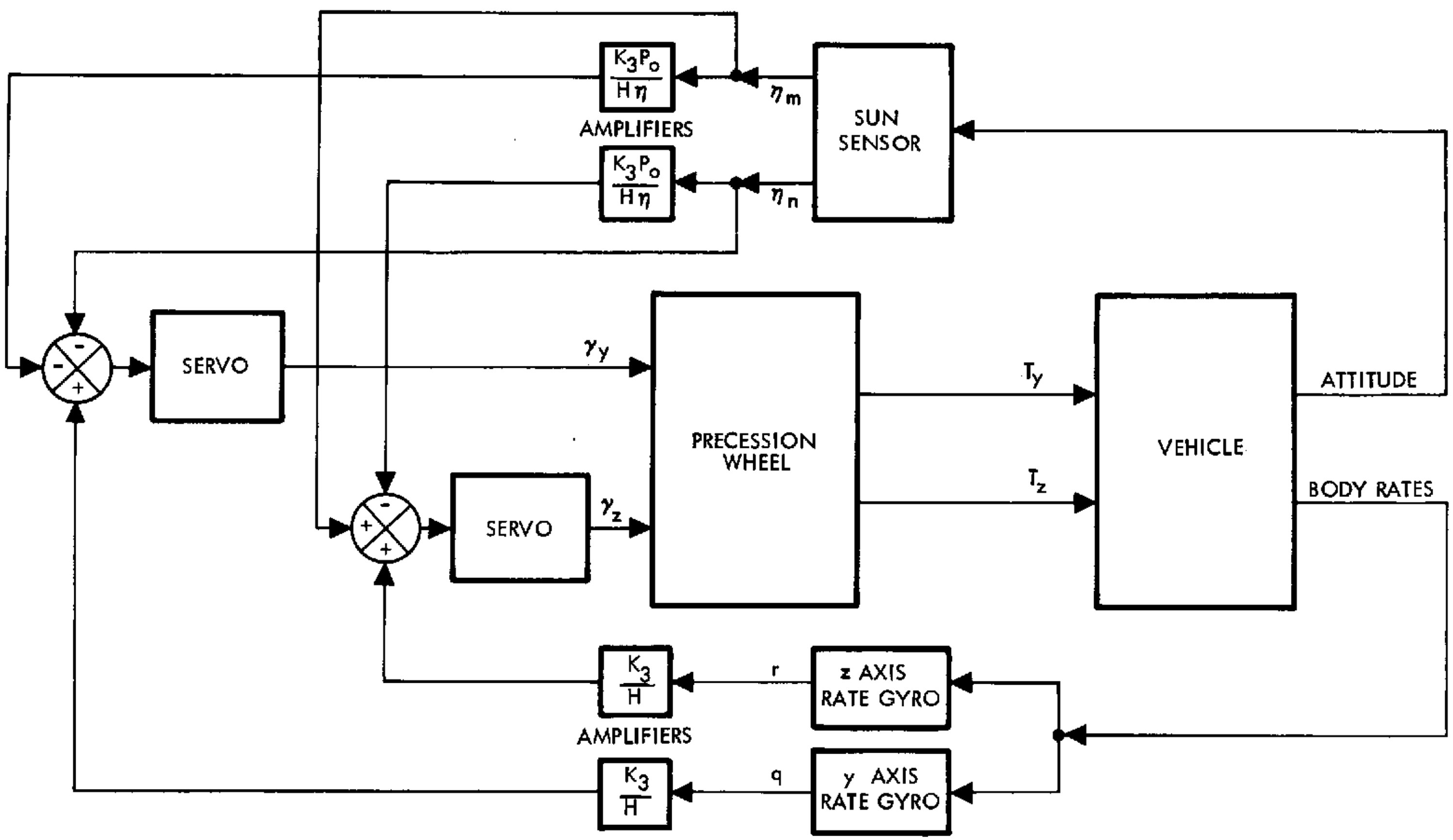


Figure 43. System Diagram - Acceleration-Controlled Precession Wheel Wobble Damper

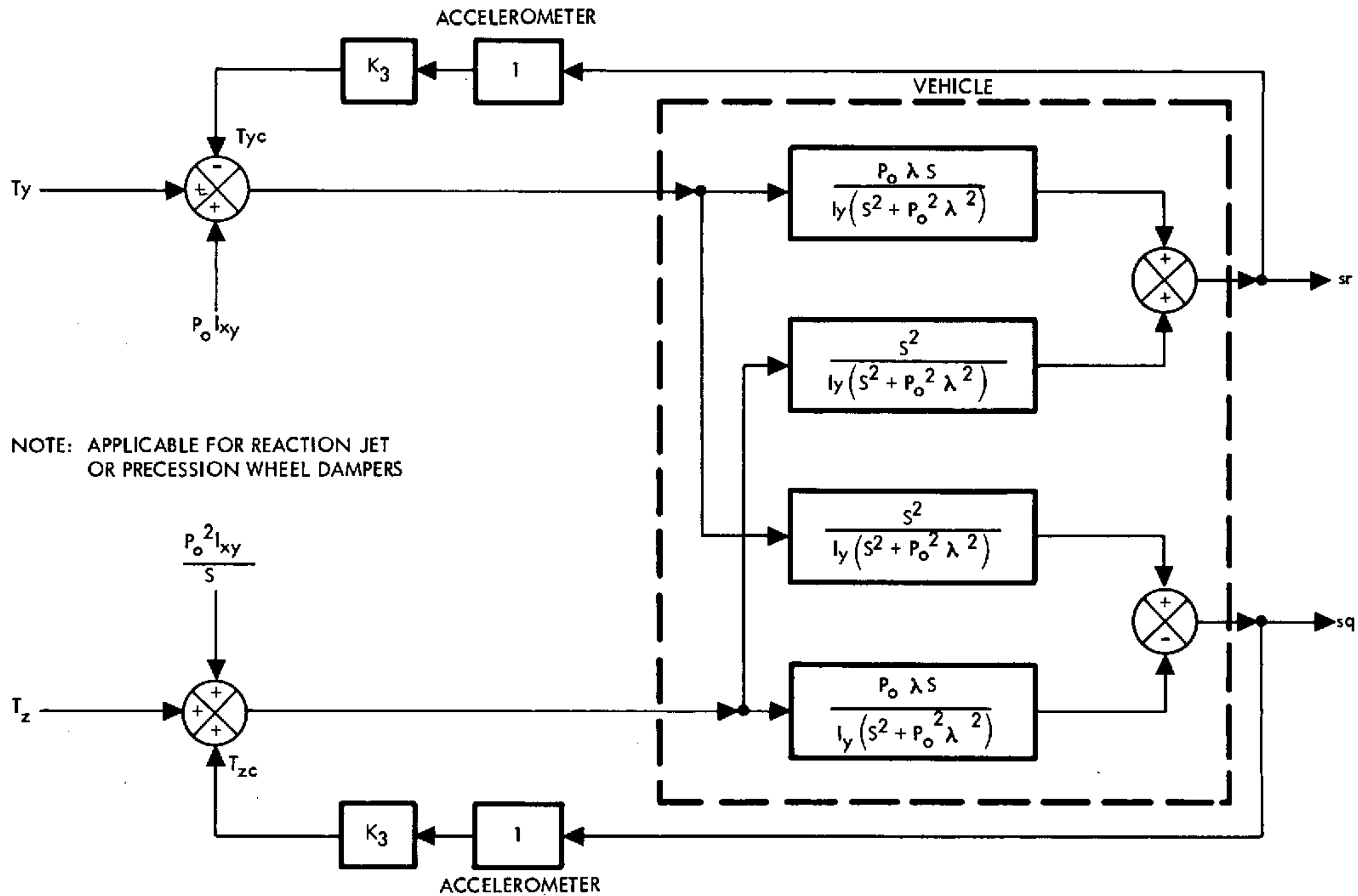


Figure 44. Block Diagram - Acceleration-Controlled Wobble Damper

If the control torques are considered to be

$$T_y = -K_3\dot{r}$$

$$T_z = K_3\dot{q}$$

then in the s domain:

$$T_y = -K_3 s r$$

$$T_z = K_3 s q$$

Setting these equal to the torques given earlier in terms of the gimbal angles:

$$-K_3 s r = -H(s\,\gamma_z + p_o\,\gamma_y + r)$$

$$K_3 s q = H(s\,\gamma_y + q - p_o\,\gamma_z)$$

Solving for γ_y and γ_z:

$$\gamma_y = \frac{K_3}{H}q - \frac{K_3 p_o}{H}\left(\frac{p_o q - sr}{s^2 + p_o^2}\right) - \left(\frac{sq + p_o r}{s^2 + p_o^2}\right) = \frac{K_3}{H}q - \frac{K_3 p_o}{H}\,m - n$$

$$\gamma_z = \frac{K_3}{H}r - \frac{K_3 p_o}{H}\left(\frac{sq + p_o r}{s^2 + p_o^2}\right) + \left(\frac{p_o q - sr}{s^2 + p_o^2}\right) = \frac{K_3}{H}r - \frac{K_3 p_o}{H}\,n + m$$

Consider an angular impulsive torque $T_y\delta\ (t)$ applied to the vehicle at t = 0 with the acceleration control laws in effect. This torque has the effect of shifting the system's total momentum by an angle equal to:

$$\frac{T_y}{p_o I_x + H} = \frac{T_y}{p_o I_y(\lambda + 1) + H}\ \text{rad.}$$

The vehicle body rates for this case are

$$q = \frac{T_y I_y s}{(I_y^2 + K_3^2)\left\{s^2 + \frac{2K_3 \lambda I_y p_o}{I_y^2 + K_3^2} s + \frac{\lambda^2 I_y^2 p_o^2}{I_y^2 + K_3^2}\right\}}$$

$$r = \frac{T_y K_3 \left(s + \frac{\lambda I_y p_o}{K_3}\right)}{(I_y^2 + K_3^2)\left\{s^2 + \frac{2K_3 \lambda I_y p_o}{I_y^2 + K_3^2} s + \frac{\lambda^2 I_y^2 p_o^2}{I_y^2 + K_3^2}\right\}}$$

The time domain solutions of these equations are damped sinusoids and the steady-state values are

$$q_{ss} = 0$$

$$r_{ss} = 0$$

Thus, wobble is completely damped in the steady state. The system time constant is

$$\frac{I_y^2 + K_3^2}{K_3 \lambda I_y p_o}$$

and the natural frequency is

$$\frac{\lambda I_y p_o}{(I_y^2 + K_3^2)^{1/2}} .$$

The attitude of the vehicle referred to the y body axis is equal to

$$m = \frac{-T_y K_3 s \left\{s - \frac{p_o I_y (1 - \lambda)}{K_3}\right\}}{(s^2 + p_o^2)(I_y^2 + K_3^2)\text{ (characteristic equation)}}$$

The inverse La Place transform of this expression is composed of a damped sinusoid and a sustained oscillation at the vehicle spin frequency of p_o. This is to be expected for the vehicle displaced from the sun line. Since the steady-state body rates are zero, the steady-state expression for n would represent a cosinusoidal variation. The gimbal angle thus oscillates at the spin frequency as shown:

$$\gamma_y = \left(\frac{K_3}{H}\right) q - \left(\frac{K_3 p_o}{H}\right) m - n$$

$$(\gamma_y)_{ss} = \frac{K_3}{H} \times 0 + A \sin(p_o t + \psi_1) + B \cos(p_o t + \psi_2)$$

$$= C \sin(p_o t + \psi_3)$$

where A, B, C, ψ_1, and ψ_2 are constants.

Since q_{ss} and r_{ss} are constant, then the magnitude of the oscillatory term in the equation for m is indicative of the final displacement of the vehicle from the sun line and is given by

$$\frac{T_y}{p_o} \left[\frac{K_3^2 + I_y^2 (\lambda - 1)^2}{I_y^4 (\lambda^2 - 1)^2 + 2K_3^2 (\lambda^2 + 1) I_y^2 + K_3^4} \right]^{1/2}$$

For a time constant equal to three and one-half spin periods the value of K_3 is 0.9×10^6. After wobble damping is accomplished, the vehicle spin axis shift is $0.1996\ T_y \times 10^{-6}$ radians from the sun line, whereas the shift in the total momentum vector is $0.2\ T_y \times 10^{-6}$ radians. Study is continuing along these lines in order to determine whether any modifications in the acceleration control laws are needed for use with the precession wheel damper.

For comparison purposes, a power estimate is developed below for an I_{xy} U(t) type disturbance using the acceleration control laws. The vehicle body rates with this control are identical to that obtained for the reaction-jet system and are repeated here for convenience.

$$q = \frac{p_o I_{xy} I_y \left\{ s^2 - \frac{K_3 p_o}{I_y} - \lambda p_o^2 \right\}}{\left\{ I_y^2 + K_3^2 \right\} \left\{ s^2 + \frac{2K_3 \lambda I_y p_o}{I_y^2 + K_3^2} s + \frac{\lambda^2 I_y^2 p_o^2}{I_y^2 + K_3^2} \right\} s}$$

$$r = \frac{K_3 p_o I_{xy}\left\{s + \dfrac{I_y I_{xy} p_o (\lambda + 1)}{K_3}\right\}}{\left\{I_y^2 + K_3^2\right\}\left\{s^2 + \dfrac{2K_3 \lambda I_y p_o}{I_y^2 + K_3^2} s + \dfrac{\lambda^2 I_y^2 p_o^2}{I_y^2 + K_3^2}\right\}}$$

The steady-state values of these rates are

$$q_{ss} = -\frac{p_o I_{xy}}{\lambda I_y}$$

$$r_{ss} = 0$$

Note that these values do not contain any control system parameters. Furthermore, they represent the rates that are obtained when the vehicle new principal axis is aligned with the total momentum. Therefore, no continuous torque is required to maintain this attitude.

Using the control law equations and assuming $K_3^2 \ll I_y^2$, the following expression for the y gimbal angle is obtained:

$$\gamma_y = \frac{I_{xy} p_o K_3}{I_y H}\left\{\frac{s - \dfrac{H}{K_3}}{s^2 + \dfrac{2K_3 \lambda p_o}{I_y} s + \lambda^2 p_o}\right\}$$

As with the previous rate control laws assume a time constant of approximately three and one-half spin periods and H = 20,000. Using the applicable space station inertias, the time domain solution is

$$\gamma_y = 0.033 \times 10^{-6} I_{xy} \frac{K_3}{H} e^{-0.017 K_3 \times 10^{-6} t} \sin(0.166t + \psi) \text{ rad}$$

where

$$\psi = \tan^{-1} \frac{0.166}{-\dfrac{H}{K_3} - 0.017 K_3 \times 10^{-6}} \text{ rad}$$

The peak gimbal angle occurs at t = 0+ and is given by

$$\gamma_y \max = 0.033\, I_{xy} \frac{K_3}{H} \times 10^{-6} = 1.5\, I_{xy} \times 10^{-6} \text{ radians}$$

Differentiating the expression given above for the gimbal angle, the gimbal angle rate is

$$\dot{\gamma}_y = 0.033\, I_{xy} \frac{K_3}{H}\, e^{-0.017K_3 \times 10^{-6} t} \left\{ -0.017K_3 \times 10^{-6} \cos 0.166t \right.$$

$$\left. - 0.166 \sin 0.166t \times 10^{-6} \text{ rad/sec} \right\}$$

The torque applied to this gimbal is

$$T_y = -K_3 sr$$

Substituting the expression for r given earlier, the time solution is obtained as follows:

$$T_y = -0.003\, I_{xy} K_3^{\,2} \left\{ \delta(t) + \frac{5 \times 10^6}{K_3}\, e^{-0.017K_3 \times 10^{-6} t} \cos 0.166t \right\}$$

$$\times 10^{-12} \text{ lb-ft}$$

The maximum torque occurs at t = 0+ and is equal to

$$T_y \max = -13,500 \times 10^{-6}\, I_{xy} \text{ lb-ft}$$

Again, a step application of product of inertia is assumed to be unrealistic and the impulsive term is ignored.

The power needed to drive this gimbal then is

$$P_y = -K_3 \dot{r} \dot{\gamma}_y \times \frac{746}{550} \text{ watts}$$

$$= A\ \delta(t) + 732 \frac{K_3^{\ 2}}{H} I_{xy}^{\ 2} e^{-0.033 K_3 \times 10^{-6} t} \cos 0.166t \left\{ 0.017 K_3 \cos 0.166t \right.$$

$$\left. + 0.166 \sin 0.166t \right\} \times 10^{-18} \text{ watts}$$

The peak occurs at t = 4 seconds and is equal to

$$P_y \max = 235.9\, I_{xy}^{\ 2} \times 10^{-12} \text{ watts}$$

Reaction-Jet Damper

As reaction jets will be used on the space station for purposes of attitude control, a wobble damping system utilizing these jets was devised for use as a back-up in case of failure in the proposed precession-wheel wobble damping system.

System Description

A system diagram of the reaction-jet damping system is shown on Figure 45. Two angular accelerometers are used to measure the vehicle angular acceleration about each of two axes which are normal to each other and both normal to the vehicle spin axis. The outputs of these accelerometers are fed through jet switching logic which generates jet firing commands. A pulse modulation technique is used to obtain a semilinear impulse versus time response from the constant thrust jets.

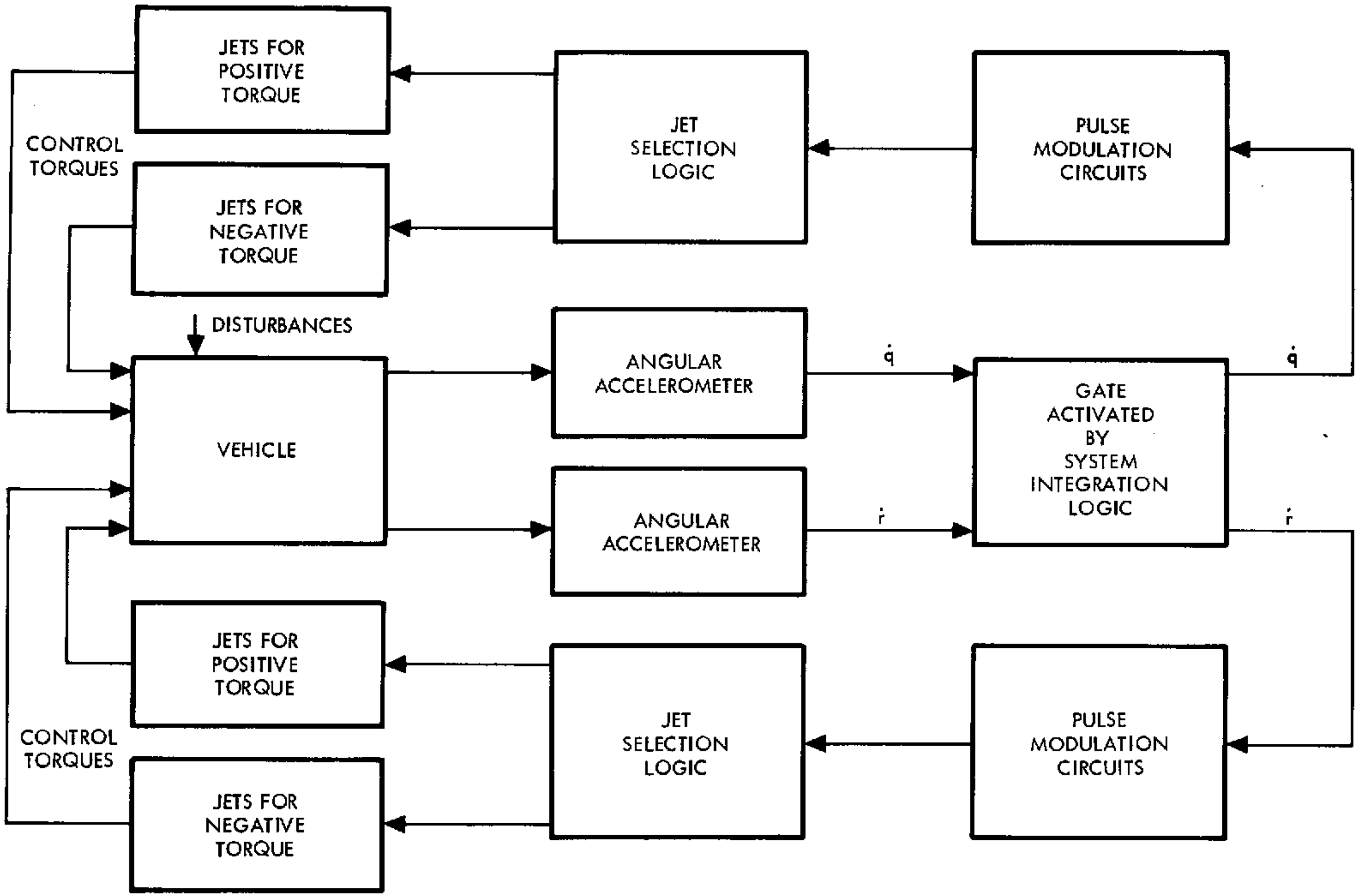


Figure 45. System Diagram - Reaction-Jet Wobble Damper

Two schemes are available for pulse modulation. The repetition rate of constant duration pulses can be varied as a function of the torque commanded or the duration of pulses occurring at a constant frequency can be varied to obtain a similar effect.

Let us first consider the scheme employing pulse repetition rate modulation. In this system, a constant duration pulse occurs each time a jet fires unless the system is saturated by a command so large that the jet remains continuously on. Below the saturation level, quasilinear impulse variations are obtained by varying the frequency at which the constant duration pulse occurs. This principle of operation is illustrated in Figure 46. The pulse width modulation scheme is also illustrated in Figure 46. In this system, the torque command is sampled and the appropriate jets are pulsed for a time duration proportional to the magnitude of torque command at the instant of sampling. The result obtained by using either of the modulation schemes is similar. Over a finite period of time the integrated effect of the torque pulses approximates the angular impulse which has been commanded.

The proposed location of reaction jets on the vehicle is sketched in Figure 47. This particular configuration was selected to maximize the effective jet lever arm. In contrast to a configuration where the lever arm for each jet is the same length, the proposed scheme has a slight advantage. Two jets are used simultaneously to provide torque about each body axis. The purpose is to produce pure rotation of the vehicle without inducing translation.

Control Laws

The problems associated with rate feedback can be overcome by employing cross-channel acceleration feedback. For a system of this type, the control laws are

$$T_{yce} = -K\dot{r}$$

$$T_{zce} = K\dot{q}$$

One major advantage of this system can be seen from the control laws. The commanded impulse decreases in magnitude as the wobble is damped; and in the limit with zero angular acceleration, the system is inactive. The system response to various disturbances has been determined and is presented in the following section.

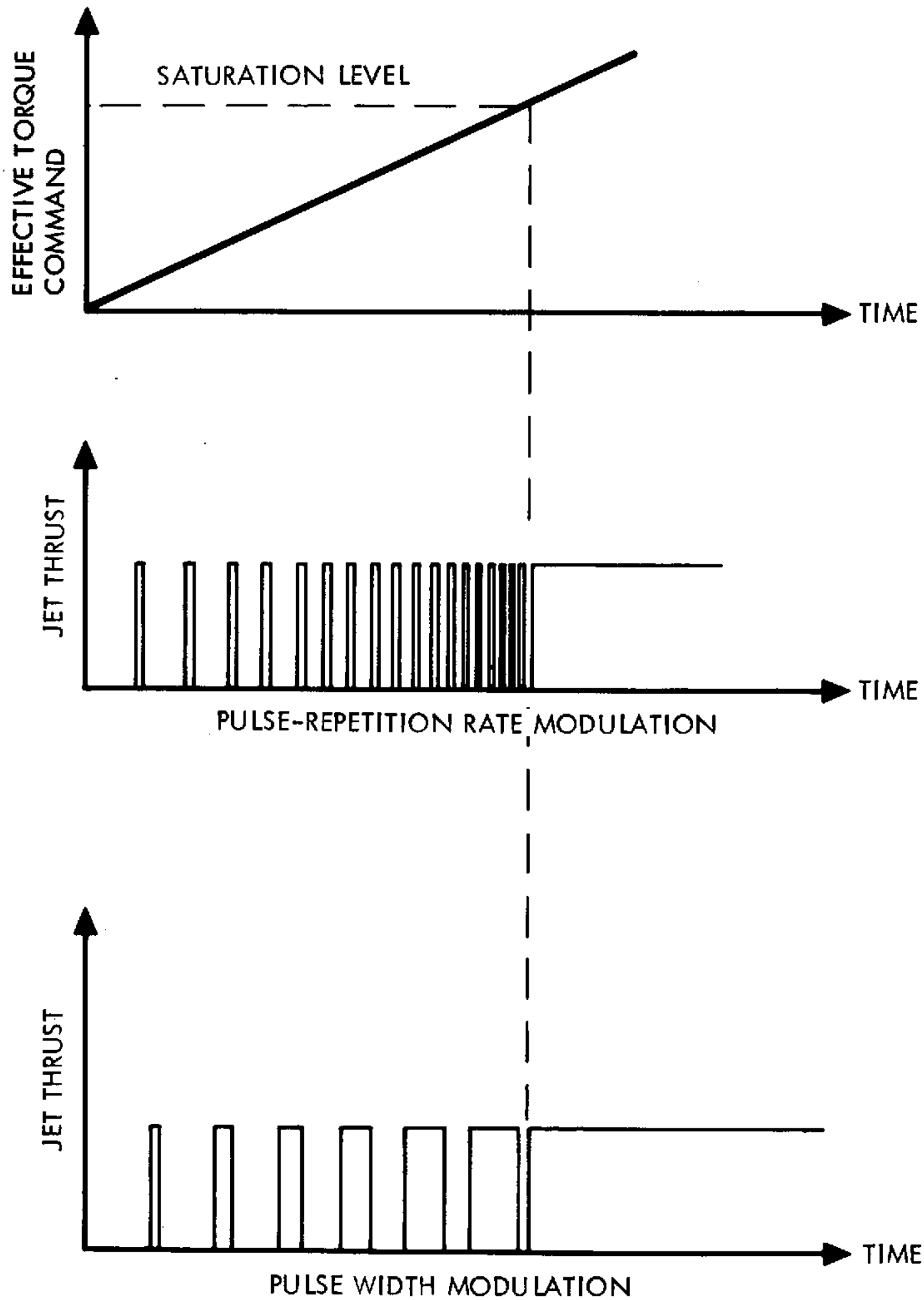


Figure 46. Reaction-Jet Pulse Modulation

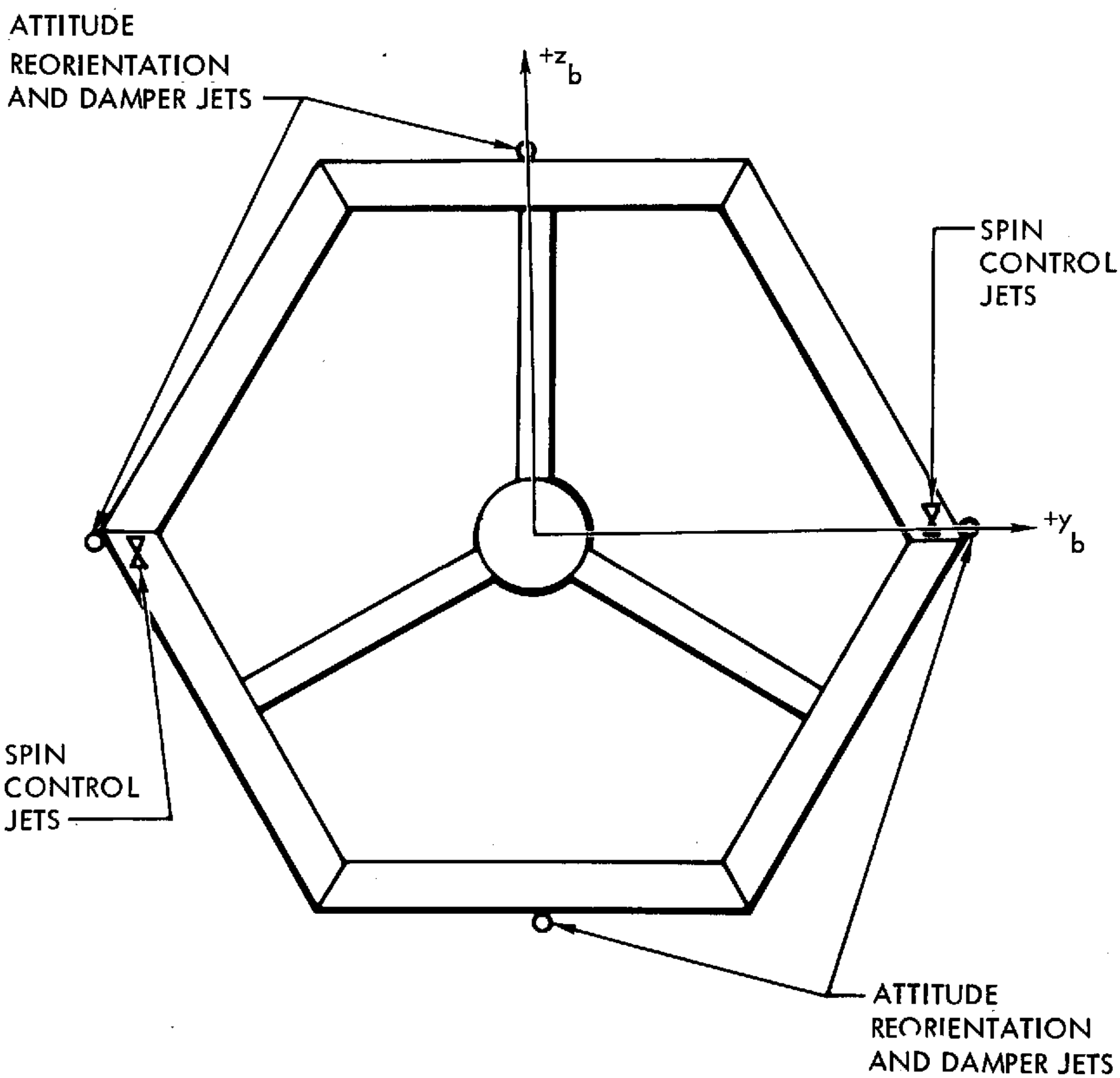


Figure 47. Proposed Reaction Jet Locations (Not Scale)

Transient Response

Consider a step product-of-inertia disturbance. The approximate transient response of the system may be determined using the linearized and simplified equations of motions previously developed. As a first approximation, the jet torques are represented by a perfect system. The equations in La Place transform notation are

$$I_y sq + p_o(I_x - I_y)r = -Ksr + p_o I_{xy}$$

$$-p_o(I_x - I_y)sq + I_y s^2 r = Ks^2 q + p_o^2 I_{xy}$$

These equations are then solved simultaneously for q and r as functions of s.

$$q(s) = \frac{-p_o I_{xy}\left[1 + \dfrac{Ks}{p_o(I_x - I_y)} - \dfrac{I_y s^2}{p_o^2(I_x - I_y)}\right]}{(I_x - I_y)\, s\left[1 + \dfrac{2K}{p_o(I_x - I_y)}\, s + \dfrac{I_y^2 + K^2}{p_o^2(I_x - I_y)^2}\, s^2\right]}$$

$$r(s) = \frac{I_{xy} I_x\left(1 + \dfrac{K}{p_o I_x}\, s\right)}{(I_x - I_y)^2\left[1 + \dfrac{2Ks}{p_o(I_x - I_y)} + \dfrac{(I_y^2 + K^2)}{p_o^2(I_x - I_y)^2}\, s^2\right]}$$

Considering the form of these functions in the s domain, it is apparent that q(t) is composed of a constant term plus a damped sinusoidal term. Application of the final value theorem will show that for steady state

$$q(t)_{ss} = \frac{-p_o I_{xy}}{(I_x - I_y)}$$

In a similar fashion r(t) is a damped sinusoid with its final value being zero.

$$r(t)_{ss} = 0$$

The damper response to a step product of inertia is therefore characterized in the steady state by constant angular rates about the body axes with the jets off.

The system response to an impulsive disturbance can be determined in a like manner and the results are summarized as follows for an impulse of magnitude J_z applied about the vehicle z body axis:

$$q(s) = \frac{-J_z\left[1 + \dfrac{K}{p_o(I_x - I_y)}\, s\right]}{p_o(I_x - I_y)\left[1 + \dfrac{2Ks}{p_o(I_x - I_y)} + \dfrac{(I_y^2 + K^2)}{p_o^2(I_x - I_y)^2}\, s^2\right]}$$

$$r(s) = \frac{J_z I_y s}{p_o^2(I_x - I_y)^2\left[1 + \dfrac{2Ks}{p_o(I_x - I_y)} + \dfrac{(I_y^2 + K^2)}{p_o^2(I_x - I_y)^2}\, s^2\right]}$$

The body rates in the time domain are both damped sinusoids with steady-state magnitudes of zero.

The system response to a constant torque disturbance was also investigated, and the results for a torque of magnitude T applied about the z body axis are

$$q(s) = \frac{-T_z\left[1 + \dfrac{K}{p_o(I_x - I_y)}\, s\right]}{p_o(I_x - I_y)\, s\left[1 + \dfrac{2K}{p_o(I_x - I_y)}\, s + \dfrac{(I_y^2 + K^2)}{p_o^2(I_x - I_y)^2}\, s^2\right]}$$

$$r(s) = \frac{I_y T_z}{p_o^2(I_x - I_y)^2\left[1 + \dfrac{2K}{p_o(I_x - I_y)}\, s + \dfrac{(I_y^2 + K^2)}{p_o^2(I_x - I_y)^2}\, s^2\right]}$$

In the time domain q(t) is composed of a constant term plus a damped sinusoid with the steady state magnitude

$$q(t)_{ss} = \frac{-T_z}{p_o(I_x - I_y)}$$

The angular rate about the z body axis, r(t), is a damped sinusoid with zero steady-state magnitude,

$$r(t)_{ss} = 0$$

With a constant torque disturbance, the damper is seen to remove the wobble and then remain inactive.

Time to Damp

One major parameter of interest in any wobble damping system is the time required to damp a disturbance or reach steady state. This time can be readily determined for each of the disturbances discussed in the previous section by analyzing the exponential decay envelope of the sinusoidal terms. The decay envelope is found to be the same for each of the three disturbances: step product of inertia, step torque, and impulse. The expression for this envelope is

$$\exp. \left\{ - \frac{p_o(I_x - I_y)K}{(I_y{}^2 + K^2)} t \right\}$$

If we consider the disturbance to be damped in three time constants, which corresponds to a 95-percent reduction in the wobble magnitude, the time to damp is

$$t = \frac{3\,(I_y{}^2 + K^2)}{p_o(I_x - I_y)K} \text{ seconds}$$

This time-to-damp expression can be written in terms of the number of spin periods required to damp υ, and in this form the expression is

$$\upsilon = \frac{3\,(I_y{}^2 + K^2)}{2\pi\, K(I_x - I_y)} \text{ spin periods}$$

This expression, if solved for K, may be used to estimate the system gain that is required to damp a disturbance in a specified number of spin periods. Figure 48 shows plots of system gain versus spin periods to damp for various combinations of vehicle inertia parameters.

Peak Thrust Required

The peak thrust required is a function of the system gain, the peak vehicle angular acceleration, and the jet lever arm. As the peak acceleration is in turn a function of the system gain, the gain must be selected first. The gain selection is made on the basis of time-to-damp considerations as discussed in the previous section. The approximate vehicle angular acceleration as a function of time for a step product-of-inertia disturbance may be obtained utilizing the equations previously developed. The resulting expression is

$$\ddot{r}(t) = \frac{I_{xy} p_o^2 (I_x^2 + K^2)^{1/2}}{(I_y^2 + K^2)^{3/2}} e^{\frac{-Kp_o(I_x - I_y)}{(I_y^2 + K^2)} t} \left[I_y \cos(\omega t + \psi) - K \sin(\omega t + \psi) \right]$$

To determine the maximum acceleration, which will occur near t = 0, the expression will be evaluated at the time corresponding to the first peak of the sinusoidal terms. Differentiating the term in brackets and setting the result equal to zero, the time at which the acceleration is maximum can be found.

$$t = \frac{\left(\tan^{-1} \frac{K}{I_y}\right) - \psi}{\omega}$$

where

$$\psi = \tan^{-1} \frac{K(I_x - I_y)}{K^2 + I_x I_y}$$

and

$$\omega = \frac{p_o I_y (I_x - I_y)}{I_y^2 + K^2}$$

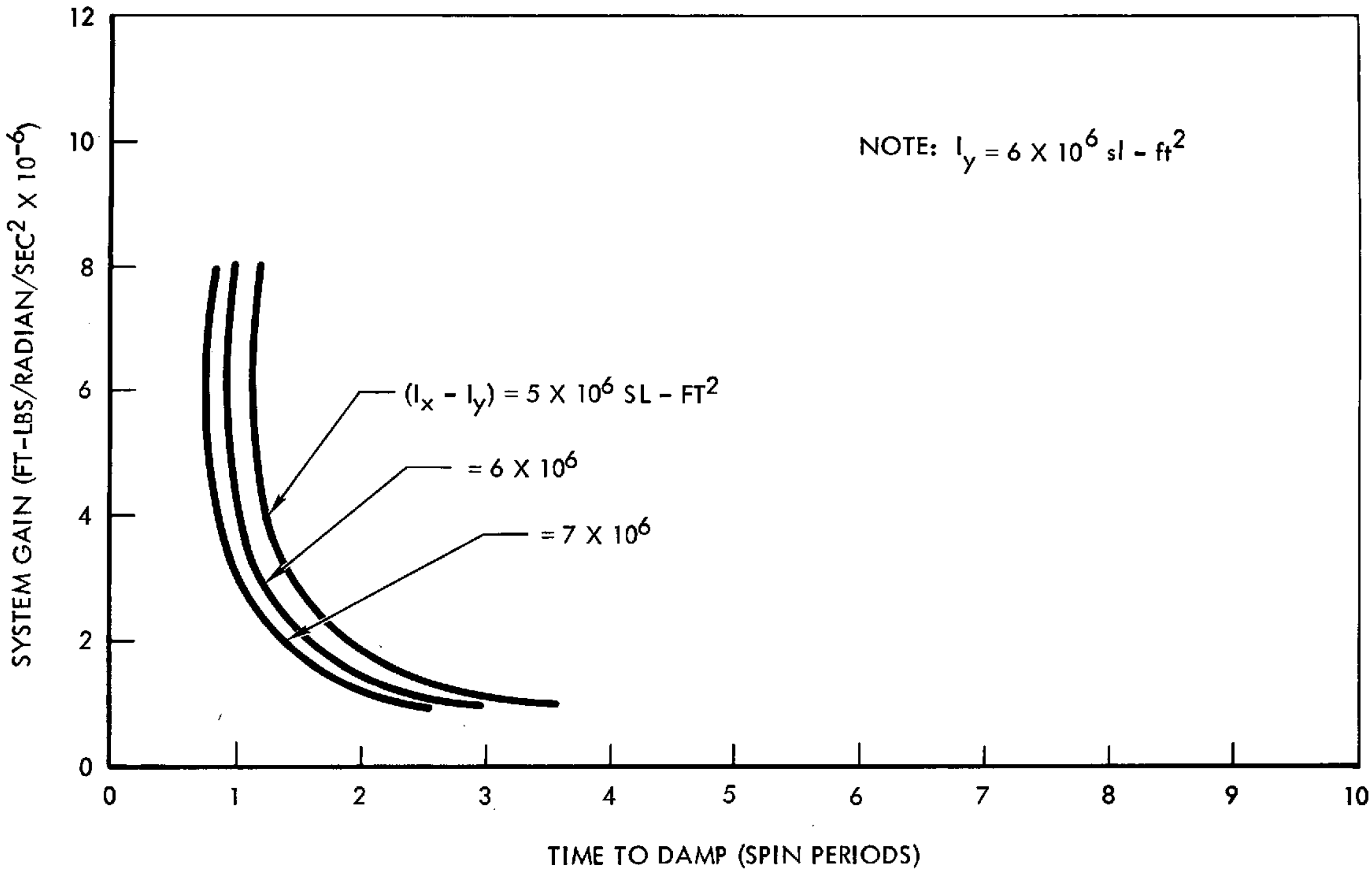


Figure 48. System Gain Versus Time to Damp for an Acceleration-Controlled Damper

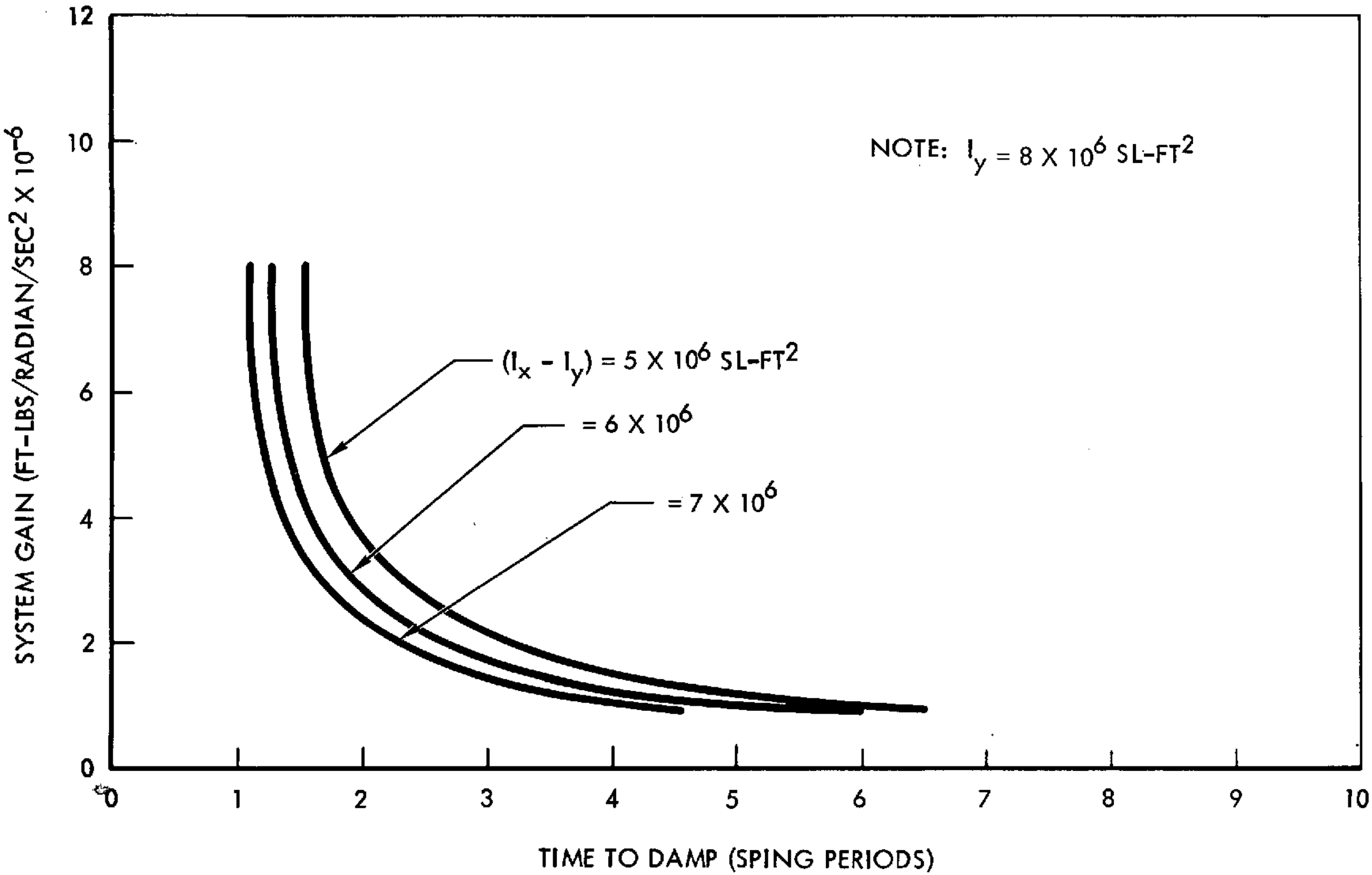


Figure 48. System Gain Versus Time To Damp For An Acceleration-Controlled Damper (Cont)

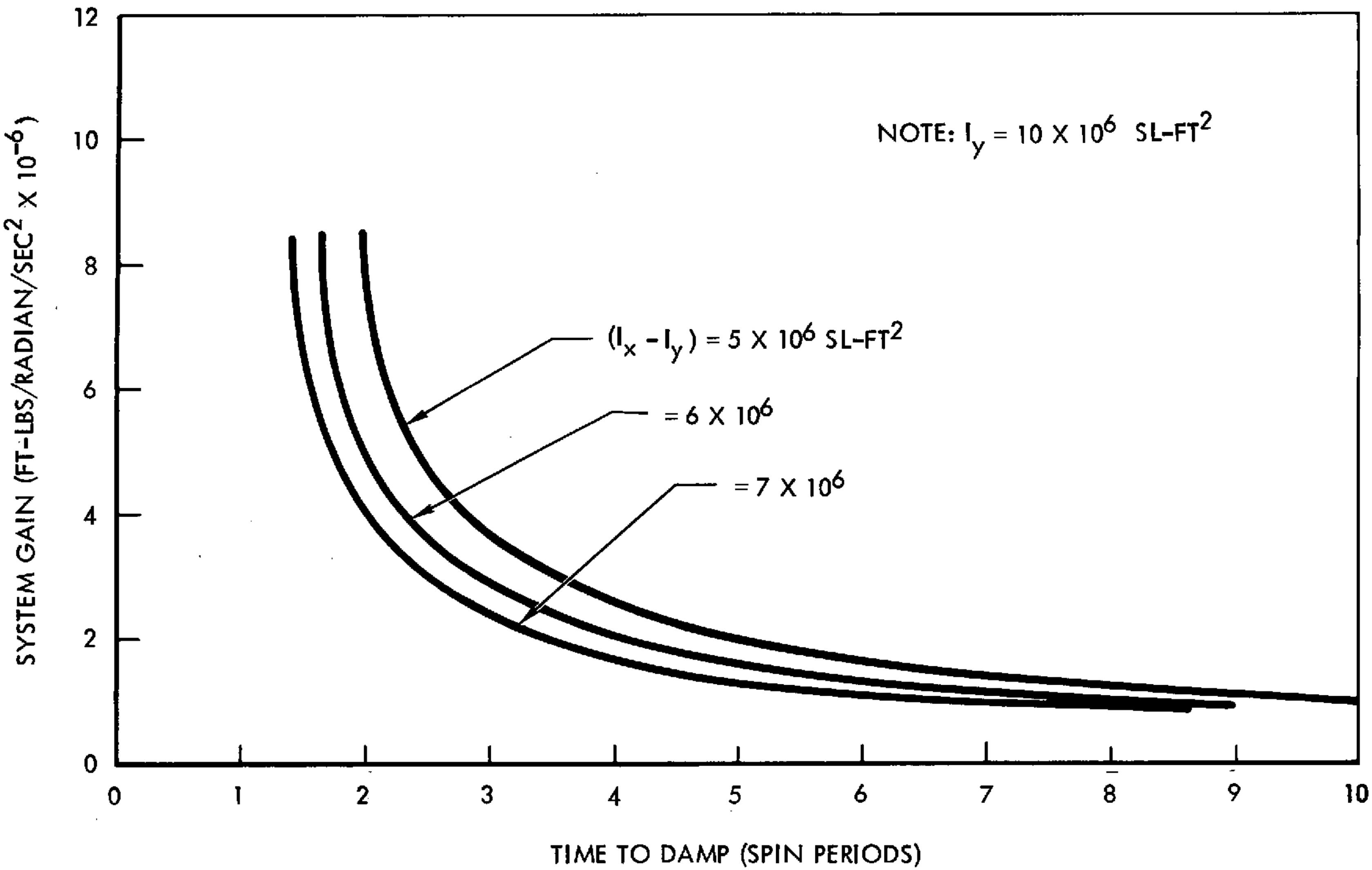


Figure 48. System Gain Versus Time To Damp For An Acceleration-Controlled Damper (Cont)

With t so defined, the expression for angular acceleration may be evaluated to obtain the peak magnitude. If the torque is applied as a couple (two jets) and the lever arm for each jet is 1, the peak thrust required is

$$\text{Thrust}_{max} = \frac{K\dot{r}_{max}}{2_1}$$

As an example of the above procedure, the peak thrust required to damp the wobble associated with a step product-of-inertia disturbance of magnitude $I_{xy} = 105,000$ sl-ft^2 in two spin periods is 15.4 lb. The vehicle parameters used in this example are

$$K = 7.4 \times 10^6 \text{ ft-lb/rad/sec}^2$$

$$I_x = 15 \times 10^6 \text{ sl-ft}^2$$

$$I_y = 10 \times 10^6 \text{ sl-ft}^2$$

$$po = 0.34 \text{ rad/sec}$$

$$1 = 65 \text{ ft}$$

Propellant Required to Damp an Impulsive Disturbance

The weight of propellant that is required to damp the wobble associated with an impulsive disturbance J_z is a function of the total angular impulse applied by the damper system which shall be denoted by J_c. The relationship is defined by the following expression:

$$W_p = \frac{J_c}{1 I_{sp}}$$

where

W_p = propellant weight ~ pounds

1 = jet lever arm ~ feet

I_{sp} = propellant specific impulse ~ $\frac{\text{lb-sec}}{\text{lb}}$

The total angular impulse applied by the damper system is the sum of the quantities applies about the y and z body axes:

$$J_c = J_{yc} + J_{zc} = \int_0^{\infty} |T_{yc}| \, dt + \int_0^{\infty} |T_{zc}| \, dt$$

where T_{yc} and T_{zc} are the damping torques.

The following linearized and simplified equations of motion are used as a first approximation:

$$I_y \, sq + p_o(I_x - I_y)r = -Ksr$$

$$-p_o(I_x - I_y)q + I_y \, sr = J_z + Ksq$$

These equations are solved to obtain expressions for q and r as functions of time and then the damping torques are simply the product of the system gain and these accelerations. The time following functions neglect an impulsive term that the system would not follow:

$$T_{yce} = \frac{-J_z \, p_o(I_x - I_y)K}{(I_y^2 + K^2)} \, e^{-\frac{K \, p_o(I_x - I_y)}{I_y^2 + K^2} t} \sin\left[\frac{p_o \, I_y(I_x - I_y)}{(I_y^2 + K^2)} t + \psi_1\right]$$

$$T_{zce} = \frac{-J_z \, p_o(I_x - I_y)K}{(I_y^2 + K^2)} \, e^{-\frac{K \, p_o \, I_x - I_y)}{I_y^2 + K^2} t} \sin\left[\frac{p_o \, I_y(I_x - I_y)}{(I_y^2 + K^2)} t + \psi_2\right]$$

where

$$\psi_1 = \tan^{-1} \frac{2K \, I_y}{(I_y^2 - K^2)}$$

$$\psi_2 = \tan^{-1} \frac{(K^2 - I_y^{\,2})}{2K\,I_y}$$

These torque expressions are then integrated to determine the required impulse. To account for the absolute magnitude requirement on the term being integrated, the integration is broken into half-wave cycles and the signs are alternated. The resulting expression is

$$J_{yc \text{ or } zc} = \frac{J\,K}{p_o(I_x - I_y)} \left\{ \alpha \frac{p_o(I_x - I_y)}{(I_y^{\,2} + K^2)} \left[-K \sin \psi - I_y \cos \psi \right] + \frac{2p_o\, I_y(I_x - I_y)\, e^{-\frac{K(\phi - \psi)}{I_y}}}{(I_y^{\,2} + K^2)\left[1 - e^{-\frac{K\pi}{I_y}}\right]} \right\}$$

where $\alpha = +1$, and $\phi = 2\pi$ for ψ in the third or fourth quadrant and $\alpha = -1$, $\phi = \pi$ for ψ in the first or second quadrant. When using the foregoing expression to compute J_{yc} J_{zc} it is important to note that the appropriate ψ must be used from the expressions for T_{yce} and T_{zce}.

As an example of the foregoing computation, the weight of propellant required to damp the wobble associated with a unit impulse disturbance (1 ft-lb-sec) is 5. 5. 3 x 10^{-5} lb. The pertinent parameter magnitudes used in this computation are

$K = 7.4 \times 10^6$ ft-lb/rad/sec^2

$p_o = 0.34$ rad/sec

$I_x = 15 \times 10^6$ sl-ft^2

$I_y = 10 \times 10^6$ sl-ft^2

$l = 70$ ft

$I_{sp} = 300$

Comparative Evaluation of the Damper Systems

A significant performance comparison of the precession wheel and reaction-jet damper systems should be based upon the effects of crew motion; because of its frequency and magnitude this is considered to be the most important disturbance which these systems will encounter. A preliminary estimate indicates that the weight of reaction-jet propellant required to damp the wobble associated with this disturbance could be as much as several thousand pounds. This estimate is somewhat pessimistic since, in reality, some statistical cancellation of the crew motion effects could be expected. The exact frequency of the disturbance is unknown at the present as is a tolerance level on the resultant wobble. Thus, a detailed trade-off study could not be conducted. Future effort in this area is recommended.

SYSTEM INTEGRATION

Regardless of whether the precession wheel or the reaction jets are used, wobble damping should precede any attitude reorientation. This is necessary for proper measurement of the attitude error. Furthermore, if the precession wheel is used to control wobble, there will be no waste of reaction-jet fuel; the reorientation system is used only to correct attitude, never to damp wobble. This concept is not too restrictive since reorientation is an infrequent operation, and a delay of one to three spin periods while wobble is being damped is not significant. The reorientation system must not only wait until wobble has been damped, but it must also decide whether it can, in fact, correct a measured attitude deviation that is too large. Consider the vehicle to be nutating symmetrically about the inertial reference axis because of an internal mass unbalance. If the resulting attitude error is judged to be excessive, the reaction-jet system should not be commanded to operate, since the jet operation could never stop. At this point, a mass balancing system would be required. However, if the peak attitude deviation is deemed too large with the vehicle total momentum vector not aligned with the sun, the reorientation system could affect some correction without requiring continuous operation.

The presence of wobble is detected by measurement of non-zero values of $|\dot{q}| + |\dot{r}|$. The signal $|\dot{m}| + |\dot{n}|$ will be used to determine whether operation of the reaction jets or the mass balancing system is dictated. Non-zero values for this signal would command the former system and if this signal vanishes the latter system would be used. The termination of the attitude reorientation mode is accomplished on the basis of attitude deadband considerations and is discussed in the reorientation section of this report. At that time, the wobble damper resumes control over the vehicle. A flow chart of this suggested logic is shown in Figure 49. It is felt that system logic might be simplified if the reorientation system controlled continuously according to the following control laws:

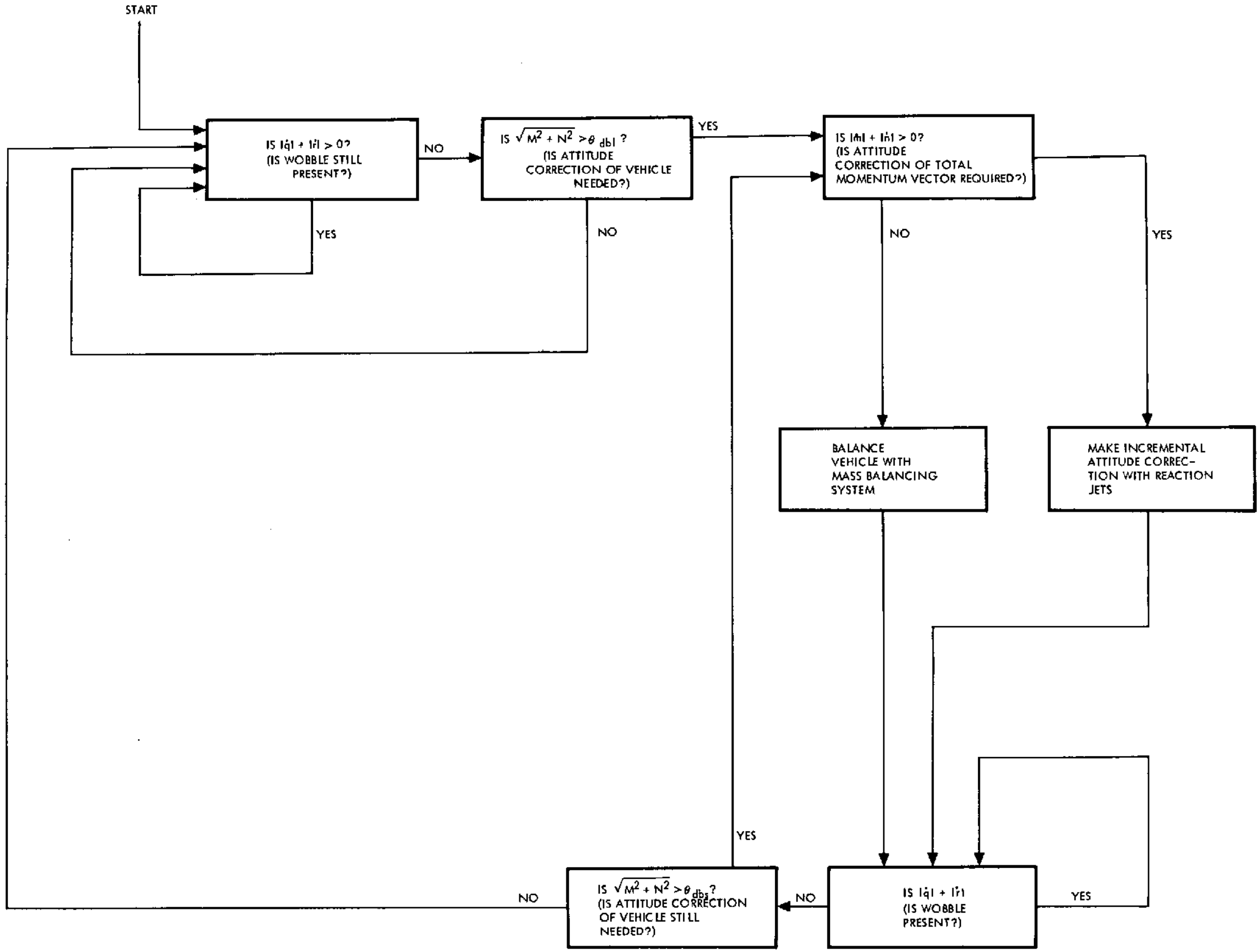


Figure 49. System Integration Flow Chart

$$T_y = -K\dot{n} = -K(q - p_o m)$$

$$T_z = K\dot{m} = K(-r + p_o n)$$

In this case, the reaction jets would automatically cut out when the vehicle spin axis and momentum vector were both aligned with the solar reference; and the system would be left at a point of stable equilibrium as previously discussed. This mechanization is presently being analyzed.

The above discussion indicates a need for several functions of of the vehicle rates and attitudes referred to the body axes. The mechanizations of these signals are shown in Figure 50. The function $|\dot{q}| + |\dot{r}|$ is obtained by rectifying the individual outputs of accelerometers located on the vehicle y and z axes. The positive outputs are then filtered and summed in a passive network to give the desired signal. The rates of change of the attitudes, $\dot{m}$ and $\dot{n}$, can be obtained by the summations $(-r + p_o n)$ and $(q - p_o m)$ respectively. In general, the vehicle spin rate is a constant; however, if it is to be varied significantly, a servomechanism could introduce the measurement from the spin control system. As with the summation of absolute values of acceleration discussed above, the function $|\dot{m}| + |\dot{n}|$ is obtained by rectifying the individual signals and adding the positive outputs. The function $\sqrt{m^2 + n^2}$ is obtained as follows: A 400 cps sine voltage is balanced modulated by the m sun sensor output. Similarly, a cosinusoidal voltage is balanced modulated by the n output. These signals are then summed and the resultant is demodulated in a conventional detector.

SPIN CONTROL

Reaction jets mounted at the rim of the vehicle and fueled with the same propellants used in the attitude control system will be used to spin up the vehicle initially and to provide any adjustment to the spin rate required throughout the mission. It is currently thought that manual activation of these jets will be satisfactory from the standpoint of reaction time required from the human operator. The expected frequency of this operation is low and the system should not require continuous monitoring by the crew. A small spin-control console will be required at the vehicle command station. This console, as a minimum, would consist of a display of either rotation rate or artificial gravity and a three-position switch corresponding to jets-on to increase the rotation rate, jets-off, and jets-on to decrease the rotation rate. The display would be fed by the output of either a rate gyro or a linear accelerometer.

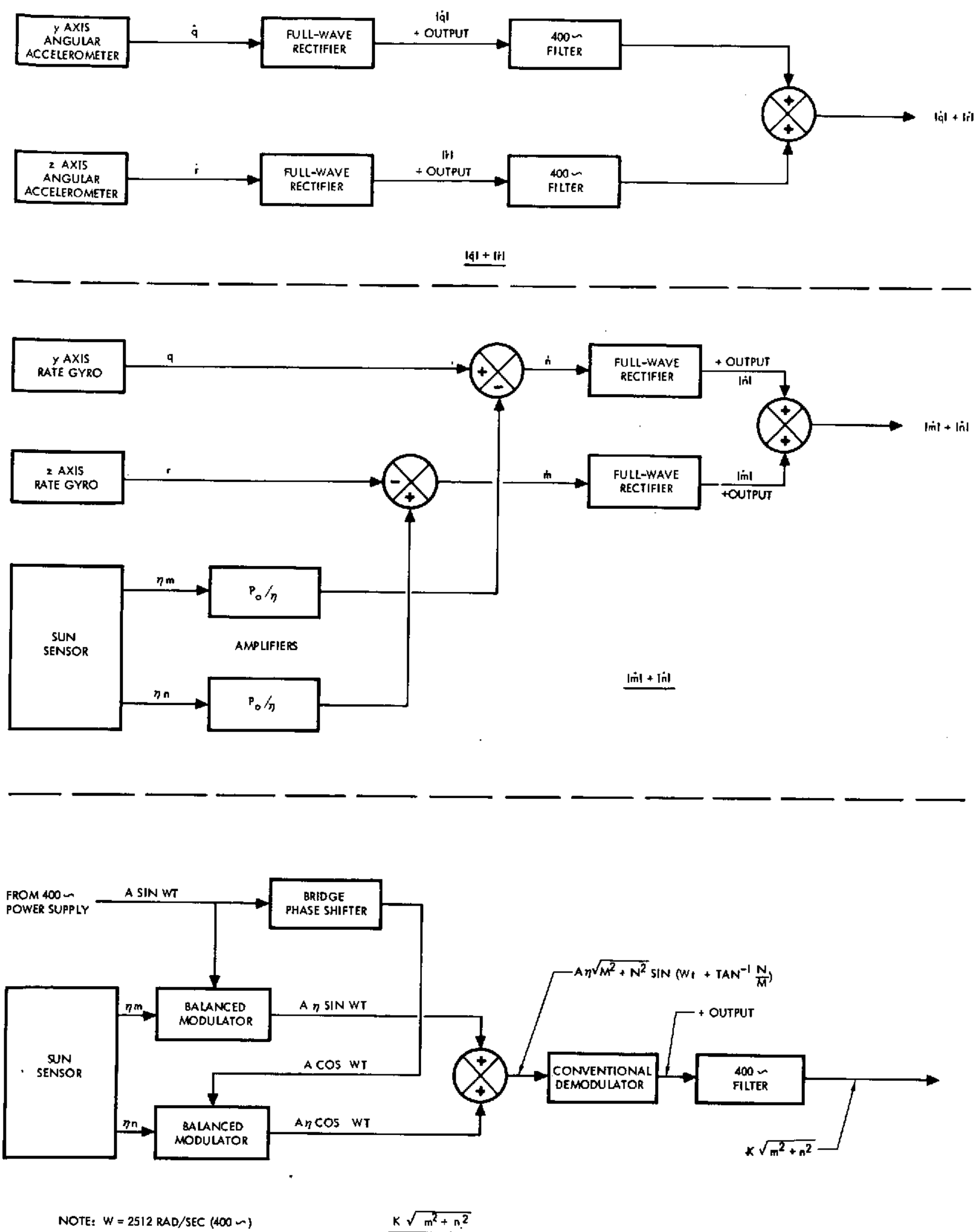


Figure 50. Logic Parameter Mechanization

If in the future, a requirement for a tight control on artificial gravity is stated, an automatic system could be employed. The development of such a system is not regarded as a major problem area. The system could be mechanized as shown in the system diagram in Figure 51.

The thrust level of the spin control jets is selected to enable accelerating to the desired rotation rate in a reasonable period of time without placing undue structural loads on the vehicle. Due to the lack of a requirement for tight control of rotation rate and the relatively minor coupling of disturbances into the spin axis, speed of response is apparently not critical. The use of 50-pound-thrust reaction jets will enable acceleration to an artificial gravity of 0.25 in less than 13 minutes. The structural loads imposed by this size jet are not critical.

The propellant required to achieve various levels of artificial gravity in the initial spin up is plotted in Figure 52.

RECOMMENDATIONS FOR FUTURE EFFORT

This section has been included to indicate, on the basis of the work completed, the areas recommended for the application of future effort. Some of the recommendations pertain to a more detailed analysis in the areas previously studied, and the remainder concern possible problem areas that were not investigated due to the time limitation.

Disturbances

Of the disturbances studied, the gravity-gradient torque represents the most severe external vehicle disturbance. A more detailed analysis of the problem is considered mandatory. The object of such a study would be to determine the effects of geometrical initial conditions, orbit inclination, and tolerable attitude error on the amount of propellant required to counter this disturbance. A digital computer analysis will probably be required.

Dynamics

There are several dynamics problems that are worthy of investigation. For example, because vehicle flexibility can be the cause of stability problems in the stabilization and control systems, a flexibility analysis should be conducted. The results of this analysis would be used in stability studies. In conjunction with the flexibility analysis, an investigation of the expected forcing functions would be required.

A second problem in the dynamics area concerns the shifting of masses on the vehicle. The transfer of a supply vehicle from the central docking port to a side port is a typical mass shift of interest. The effects of

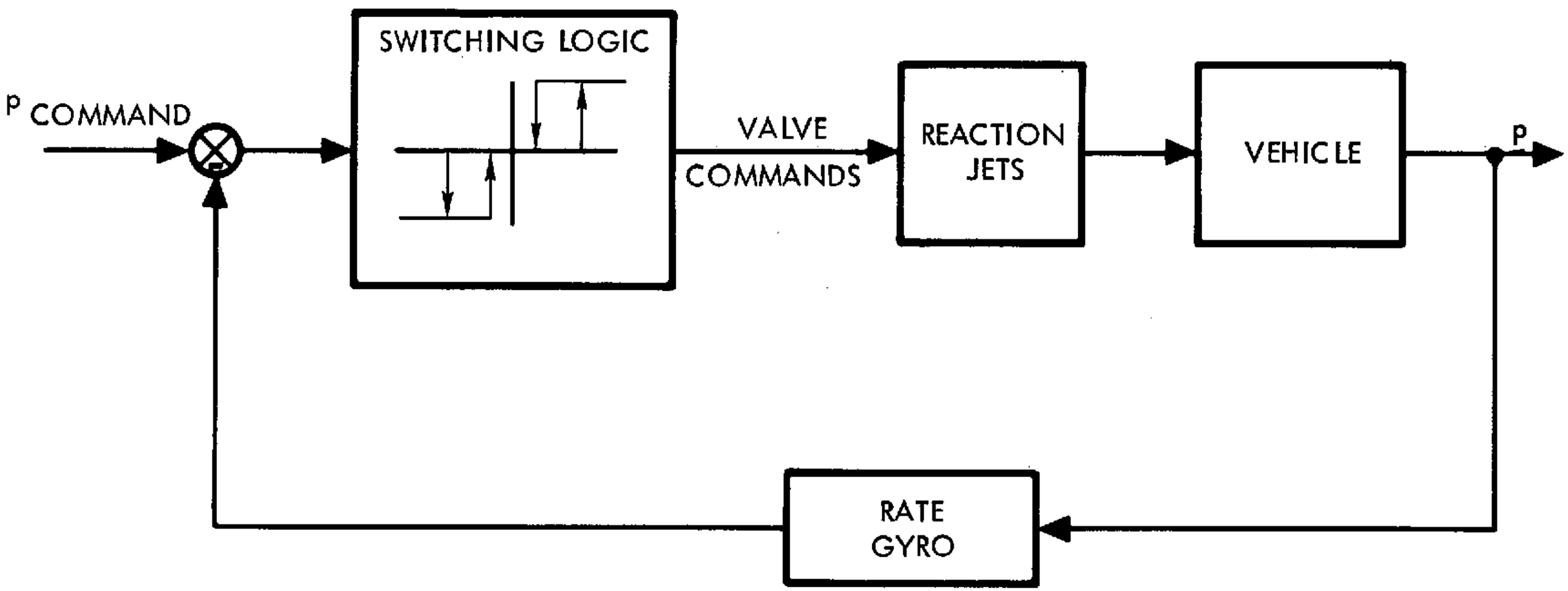


Figure 51. System Diagram - Automatic Spin Control System

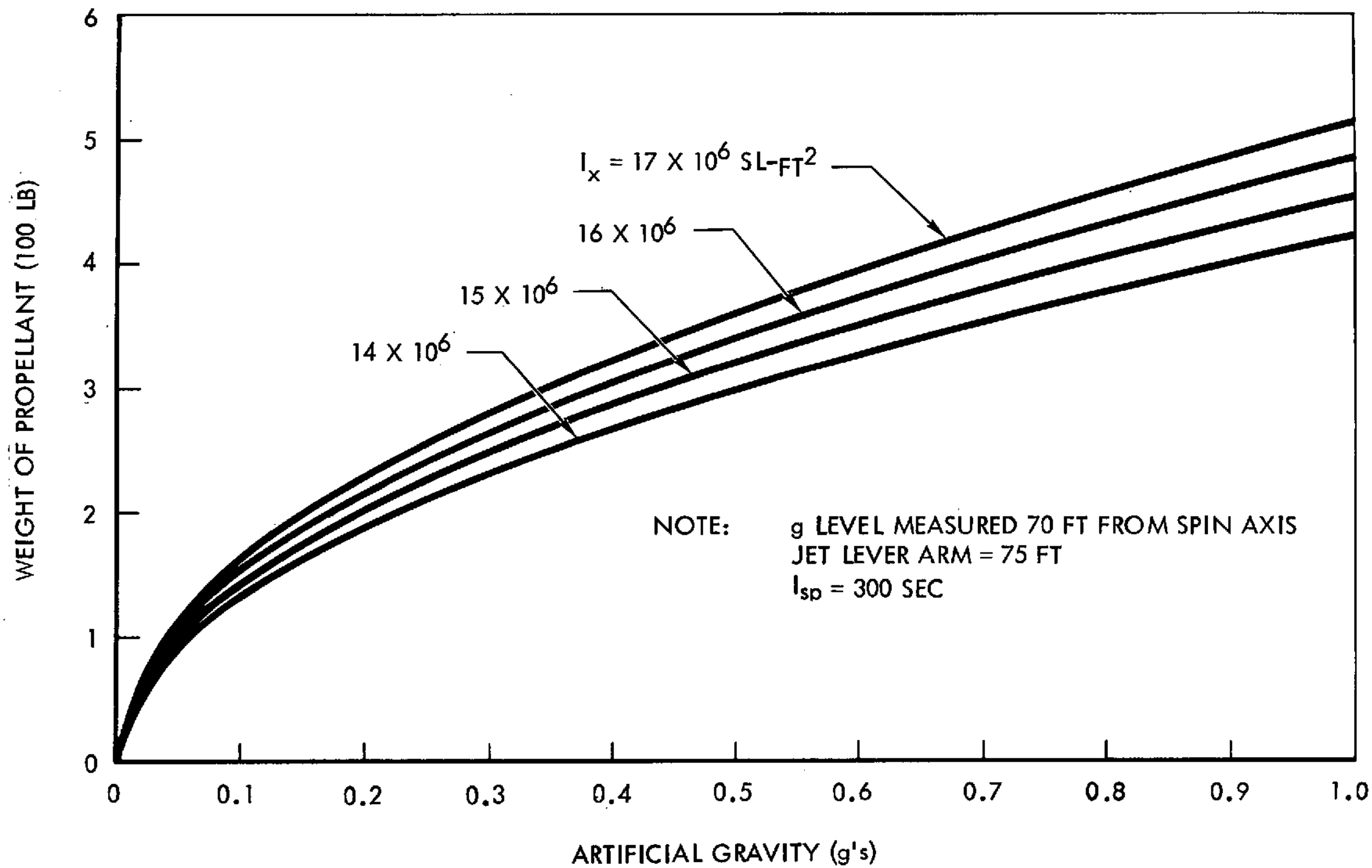


Figure 52. Weight of Reaction Jet Propellant Required to Produce Various Levels of Artificial Gravity

step-mass shift need to be correlated with continuous movement. The analysis of this problem will require a computer study. In order to specify requirements for the stowing carriage de-spin system, the above study should include the effects of imperfect cancellation of the vehicle spin rate.

Control Systems

A study will be required to determine the best location for the sensors required by the various control systems. The problems involved in such a study include structural misalignments, effects of vehicle flexibility, and noise versus path length for transmission of the output signals. Parametric studies will be required to determine hardware specifications, such as response time requirements for the reaction jets and sensor characteristics.

An important problem that should be analyzed in further detail is the integration of the attitude control and wobble-damping systems. The basic logic for such integration is presented in the report; however, it is believed that an analog computer study could effect some simplification. The study would include the mechanization of each of the systems and the associated system integration logic.

More detailed analyses of the reaction jet control systems are required to determine the effects of finite torque application time and phasing errors between adjacent jets.

The precession wheel gimbals were assumed to be controlled by perfect servomechanisms. The effects of finite bandwidths on the damper system should be investigated.

ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEMS

The general study objective in this technical area was to define environmental control and life support systems that could satisfy the space station requirements and be compatible with the configuration and vehicle systems described in other sections of this report. The study was divided into those three categories—thermal control, atmospheric control, and life support—which combined must provide a suitable working and living environment for crew members of the space station.

A number of ground rules were established during the study that were very important in the design of the environmental control and life support systems. Several ground rules were founded on the conservative design approach taken, while others were based on agreements between S&ID and LRC. The crew size of initial space station configurations was limited to six men but subsequent increases in volume and allowable weight made it possible for a larger number of crewmen in the vehicle. On the basis of discussions with NASA, a crew size of 21 men was selected as a yardstick in sizing the design of the environmental and life support systems. Another important ground rule was provision for a shirtsleeve environment, with pressure throughout the vehicle equalized to permit freedom of movement between modules. Since early studies indicated that the likelihood of a space station operational date of 1966 was feasible from all standpoints, one of the study goals was to design the environmental control and life support systems to be compatible with the state of the art in that time period. This generally indicates that current techniques in equipment design must be used to produce highly reliable systems in 1966. All equipment will be designed for easy maintenance. Malfunctions will be corrected by replacing smaller units or modules with on-board spares. For this analysis, it was assumed that expendable items would be resupplied at a minimum of 6-week intervals.

One of the first problems encountered in the conception of the environmental control system was whether to use a single, centrally-located system for the entire vehicle or to provide a smaller, but independent system in each module. A single system offers the advantage of fewer items of equipment and possibly easier maintenance; however, a major malfunction in such a system could result in complete system failure and mission abort, unnecessarily jeopardizing the lives of the crew. With separate environmental control system in the hub and in each of the six modules, a malfunction in one unit would require the occupants of that unit to move to another module until repairs have been completed. If the damage is irreparable, the module could be sealed off from the rest of the station by pressure-tight airlocks, with no

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great decrease in the performance capability of the space station. Consequently, the safety feature of redundancy provided by individual systems in the hub and in each module eliminated the concept of a centrally-located system for the entire vehicle.

Design details of equipment required for environmental control and life support functions are not presented here. The S&ID studies were basically aimed at establishing design criteria, formulating an approach to the system, and establishing the feasibility of selected concepts of the system. While in most cases these studies did not result in equipment design, the preliminary design of equipment capable of performing the required tasks emerged from requirements defined in the S&ID feasibility studies and revealed to several companies. The Hamilton-Standard Division of United Aircraft conducted a very comprehensive analysis of the criteria and equipment required for the thermal and atmospheric control systems. United's findings are presented in a supplemental report. Whirlpool Corporation, in conjunction with the S&ID analyses, has conducted extensive studies of personal hygiene requirements and food preparation and storage facilities for the space station. A supplemental report containing the details of Whirlpool's studies is being submitted separately to NASA. Similarly, preliminary equipment designs for water recovery units—designs contributed by Ionics, American Machine and Foundry, and Hamilton-Standard—are also presented in supplemental reports.

DETAILED DESIGN REQUIREMENTS

Detailed design requirements for all of the various environmental control functions are presented in Table 10, which includes all of the basic operating parameters necessary to conduct analyses of the system and to determine resupply requirements.

The metabolic balances were established by reviewing published literature and confirming these with experimental results obtained from a 14-day sealed cabin test performed by North American Aviation. The volume of the modules and hub are based on the final configuration described in "Configuration Analysis." A comparative analysis was conducted of systems operating at a total pressure of 7, 10, and 14.7 psia. On the basis of physiological factors, leakage rates, and structural considerations, a total pressure of 10 psia is recommended for the space station design.

The total space station surface area is approximately 24,000 square feet. Based on the Apollo leakage rate of 0.48 lb/day at 7 psia internal pressure for a surface area of 1330 square feet, the comparable leakage rate of the space station is on the order of 12 lb/day at 10 psia. It is not believed such a comparison is realistic, however, because of the different construction techniques employed. The space station structure—a welded shell with rigid

Table 10. Specification Requirements

Environmental Control Function	Requirement
Metabolic Consumption:	
1. Dry Food	1.40 lb/man-day
2. Oxygen	2.00 lb/man-day
3. Water	
a. Food Preparation	2.50 lb/man-day
b. Drinking	4.00 lb/man-day
c. Metabolic	1.05 lb/man-day
d. Washing	8.00 lb/man-day
Metabolic Output:	
1. Carbon Dioxide	2.25 lb/man-day
2. Water	
a. Respiration and Perspiration	4.12 lb/man-day
b. Urine	3.20 lb/man-day
c. Fecal	0.22 lb/man-day
d. Metabolic	1.05 lb/man-day
3. Feces (solid)	0.11 lb/man-day
Resupply Frequency	6 weeks minimum
Power Penalty	333 lb/kw
Module Volume:	
1. Outer Module	5,700 ft^3/module
2. Center Hub	2,500 ft^3
Atmospheric Composition:	
1. Total Pressure	10 psia
2. Partial Pressure of O_2 (minimum)	160 mm Hg

Table 10. Specification Requirements (Continued)

Environmental Control Function	Requirement
Atmospheric Composition (Continued)	
3. Partial Pressure of N_2	240 mm Hg
4. Partial Pressure of CO_2 (maximum)	7.6 mm Hg
5. Relative Humidity	50 percent
Module Temperature:	
1. Occupied Areas	75 F ± 5 F
2. Electrical Equipment	150 F maximum
Heat Rejection Requirements:	
1. Module Electrical Load Less ECS Power	2 kw
2. Hub Electrical Load Less ECS Power	1 kw
3. Module Metabolic Load	3,000 Btu/hr
4. Hub Metabolic Load	2,000 Btu/hr
Leakage Rate	10 percent/month
Repressurization Requirements:	
1. Repressurizations per Module at Launch	2
2. Repressurizations per Module at 1 Year	1

joints sealed after deployment—should reduce the specific leakage. One principal source of leakage, meteoroids penetrating the inner pressure wall, can be minimized by arranging equipment for easy access to the module's internal surface to repair the punctures. For the purpose of this analysis, a leakage rate of 10 percent per month has been assumed.

The environmental control system was designed to hold contaminants and odors at a level producing no objectionable effects on crew members. A detailed discussion of contaminant removal equipment is provided later in this section.

Before the space station is launched into orbit, each of the modules and the hub are sealed so they contain air at 14.7 psia for the initial pressurization of the spokes. Since there is a fairly high probability of accidental air loss during the deployment and initial operation aboard the space station, sufficient oxygen and nitrogen are stored in high-pressure bottles to completely repressurize the vehicle twice. Additional oxygen and nitrogen in cryogenic form are carried to compensate for leakage.

THERMAL ANALYSIS

The primary objective of the thermal control subsystem is to maintain a constant temperature in the vehicle by balancing equipment and crew heat loads with the external vehicle heat loads. Thus, it is necessary to establish the values of these heat loads. Although the thermal subsystem is integrated with the atmospheric control subsystem, it will be discussed separately.

Equipment and Crew Heat Loads

Analyses of power requirements of the vehicle equipment and of the atmospheric control system indicated that the average equipment heat load per module is 2 kw. The crew metabolic heat load was taken as 500 Btu/ man-hour, or a total of 3000 Btu per hour for six men.

External Heat Loads

External heat loads arise from three sources: direct and reflected solar radiation and earth emission. They compose the external radiant heat input. In addition, the vehicle itself radiates heat from its surface to deep space in proportion to its surface temperature.

The radiant heat input; surface temperatures; and, therefore, the heat re-radiated to space, continually change as the vehicle proceeds around its orbit. The heat inputs and surface temperatures for the surface of one module were determined by using an IBM 7090 program designed for this purpose. Orbit variables used were 300-mile altitude, 33-degree inclination to the earth's equator, and zero eccentricity. Because the space laboratory rotates to provide artificial gravity, a "spinning" effect is imparted to each module; that is, the module appears to spin about an axis perpendicular to its longitudinal axis. The IBM program was modified to account for this spinning effect.

The module surface was divided into 12 flat sides to simulate the actual cylindrical shape, as shown in Figure 53, because of the nature of the computer program. Parametric data were fed into the program, using different values of absorptivity and emissivity, resulting in the typical curves shown in Figures 54 and 55.

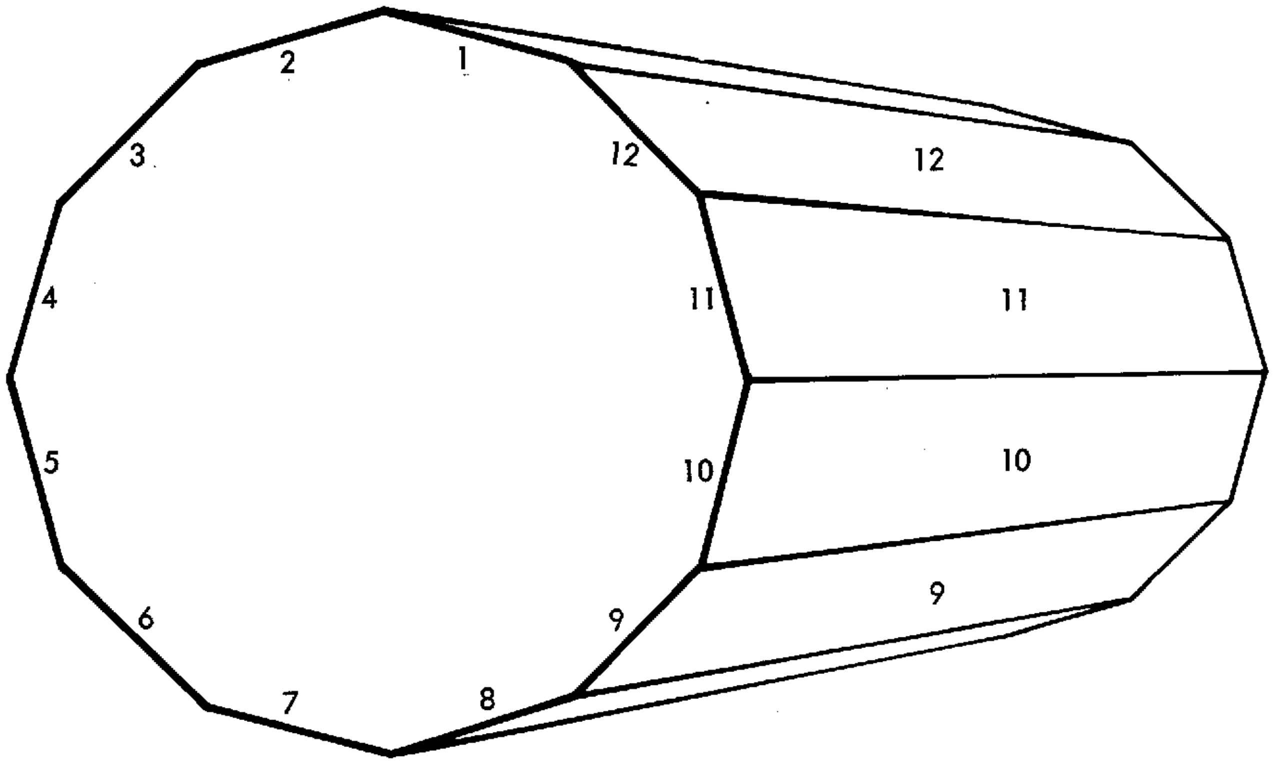


Figure 53. Thermal Analysis Model

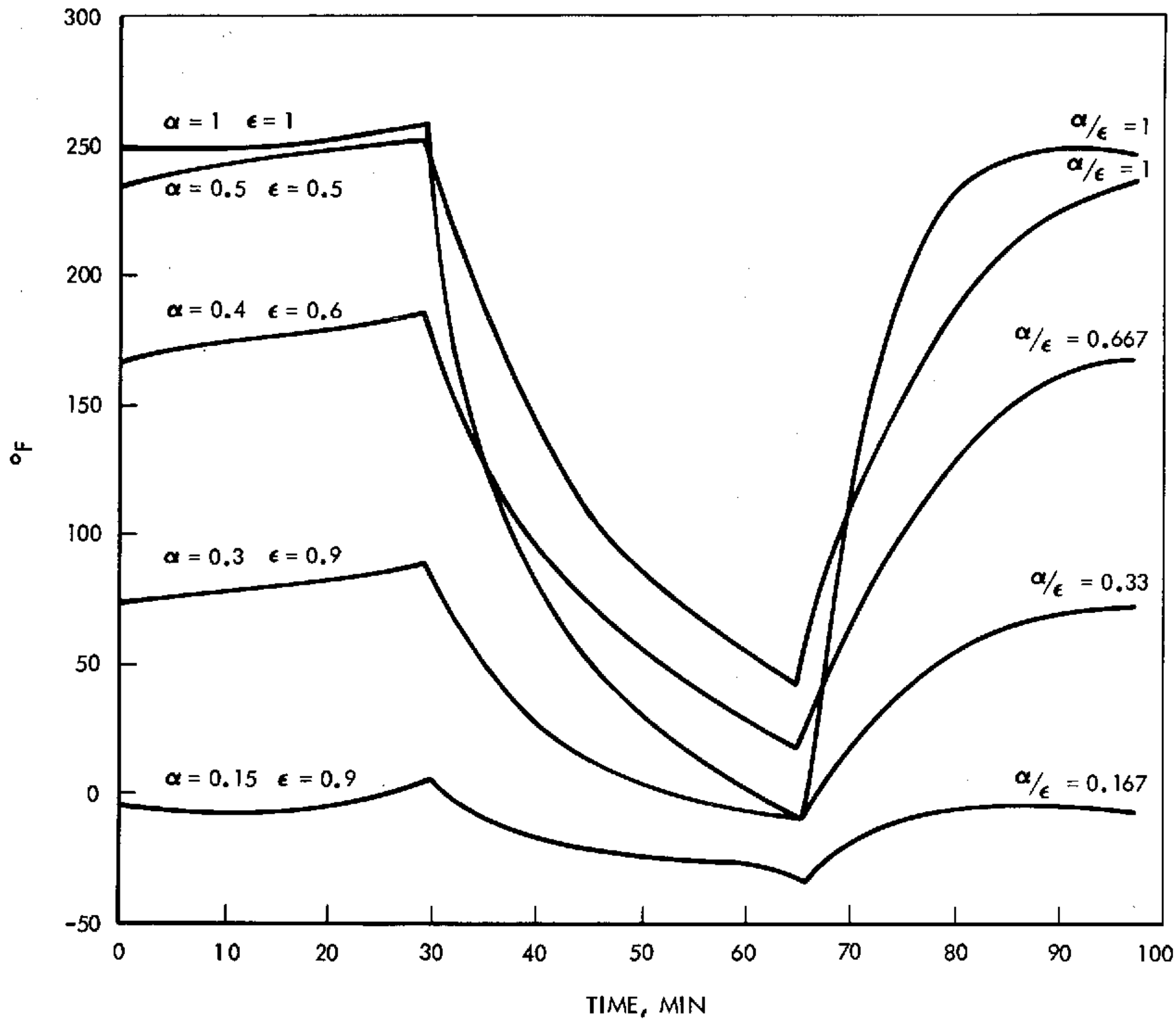


Figure 54. Typical Module Temperatures

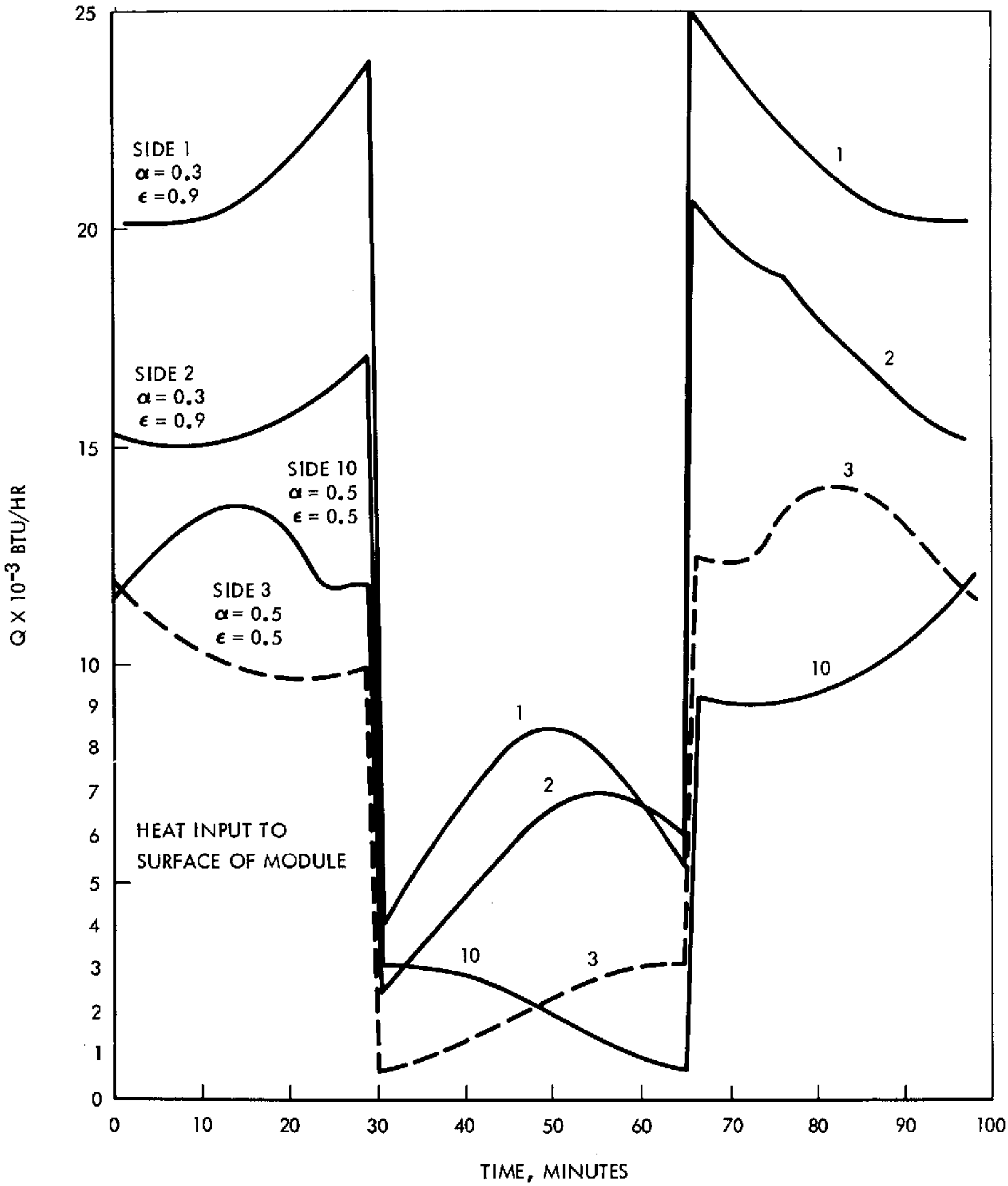


Figure 55. Heat Input to Surface of Module

From the temperature curves, it was possible to select values of absorptivity and emissivity for each of the 12 sides so that the surface temperatures would be nearly the same over the module periphery. These values were as follows:

Side Number	Absorptivity	Emissivity
1	0.3	0.9
2	0.3	0.9
3	0.5	0.5
4	0.4	0.1
5	0.3	0.1
6	0.3	0.1
7	0.3	0.1
8	0.3	0.1
9	0.4	0.1
10	0.5	0.5
11	0.3	0.9
12	0.3	0.9

Results of the analysis of the gradient through the wall showed that sides 4 and 9 should be changed to absorptivity of 0.5 and emissivity of 0.1. The computer program was re-run, using a new configuration factor subroutine, and including the module spin effects. Results of the heat inputs are shown in Figures 56 through 67, with the following nomenclature:

S - solar incident radiation

R - incident solar reflection

E - incident earth emission

A - net radiation absorbed

Temperature profiles accompanying these heat inputs are shown in Figures 68 and 69. For comparison, Figures 70 and 71 show how the temperatures vary if the module spin effect is neglected. From Figures 68 and 69, it can be noted that inclusion of the spin imparts a symmetry to the corresponding curves (i.e., sides 1 and 12, sides 2 and 11, etc.).

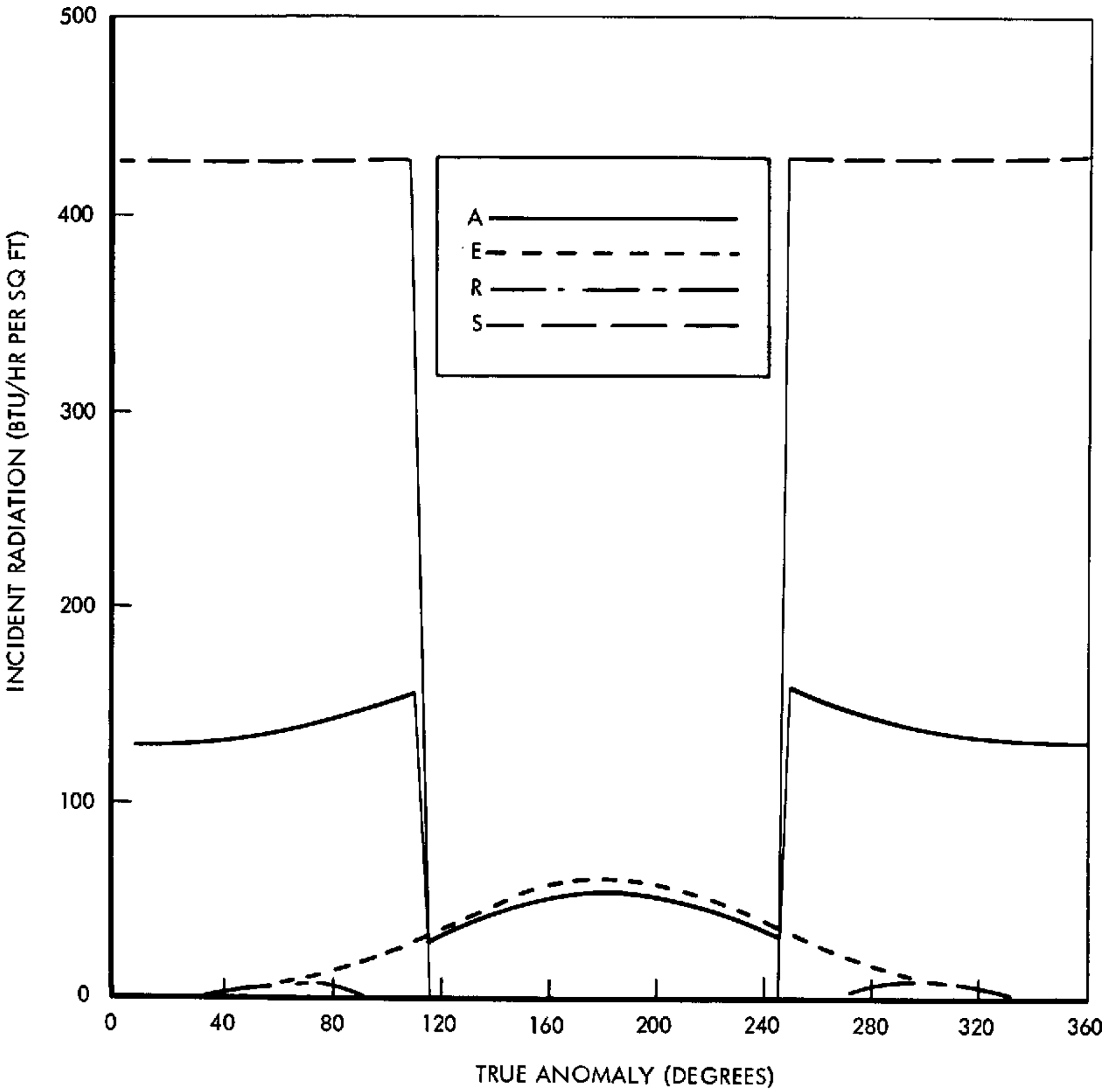


Figure 56. External Heat Load With Module Spin - Side 1

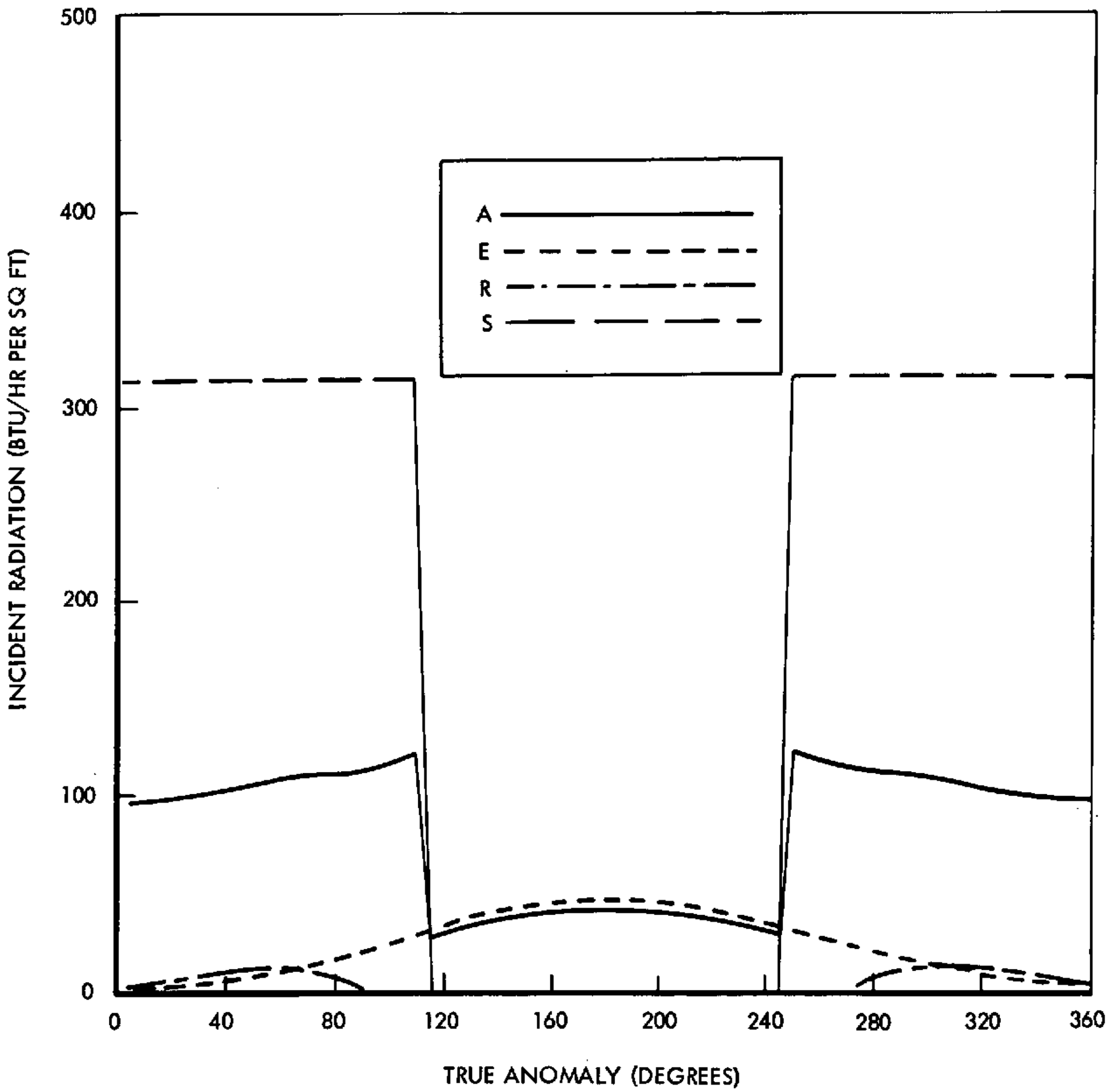


Figure 57. External Heat Load With Module Spin - Side 2

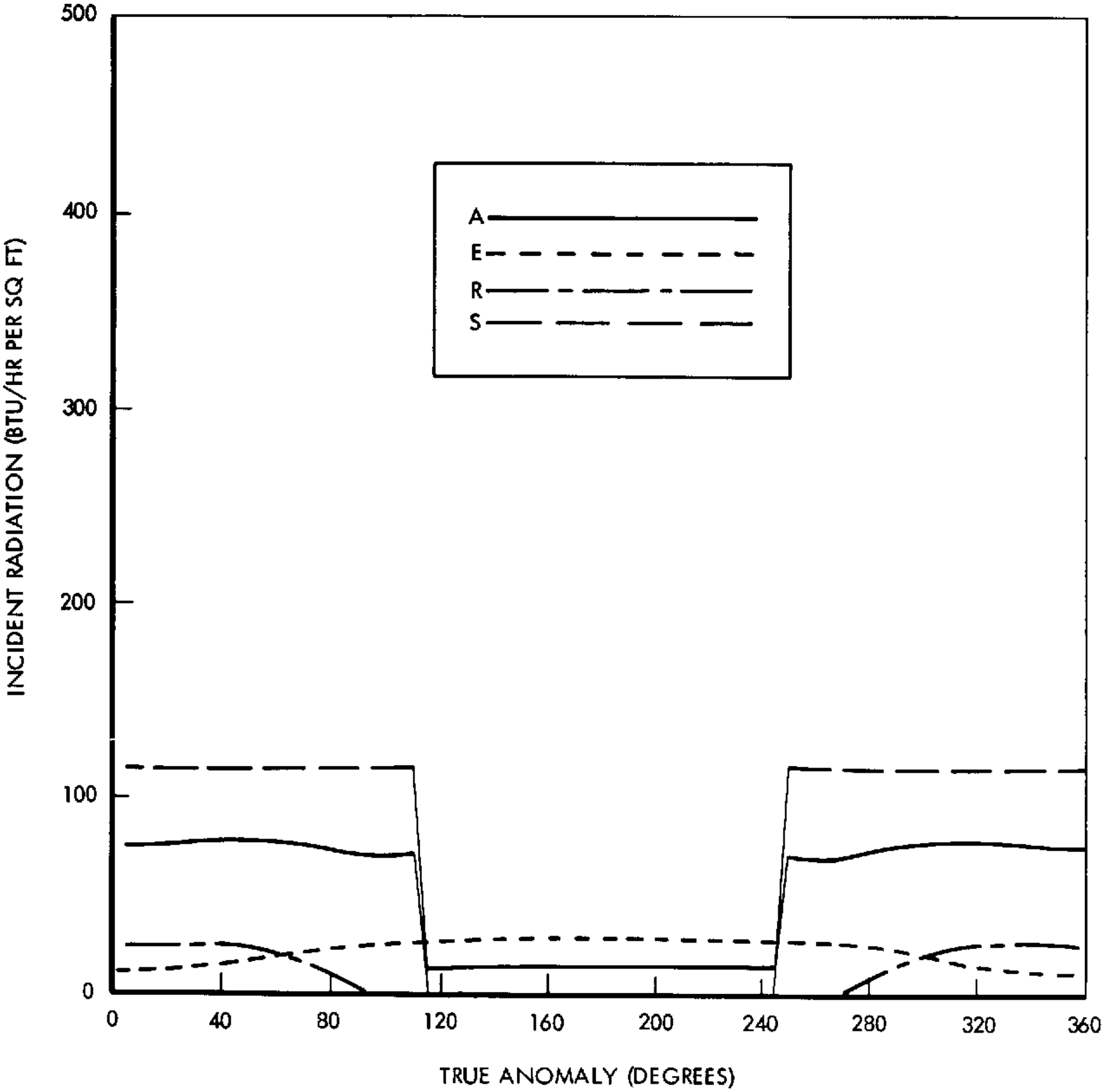


Figure 58. External Heat Load With Module Spin - Side 3

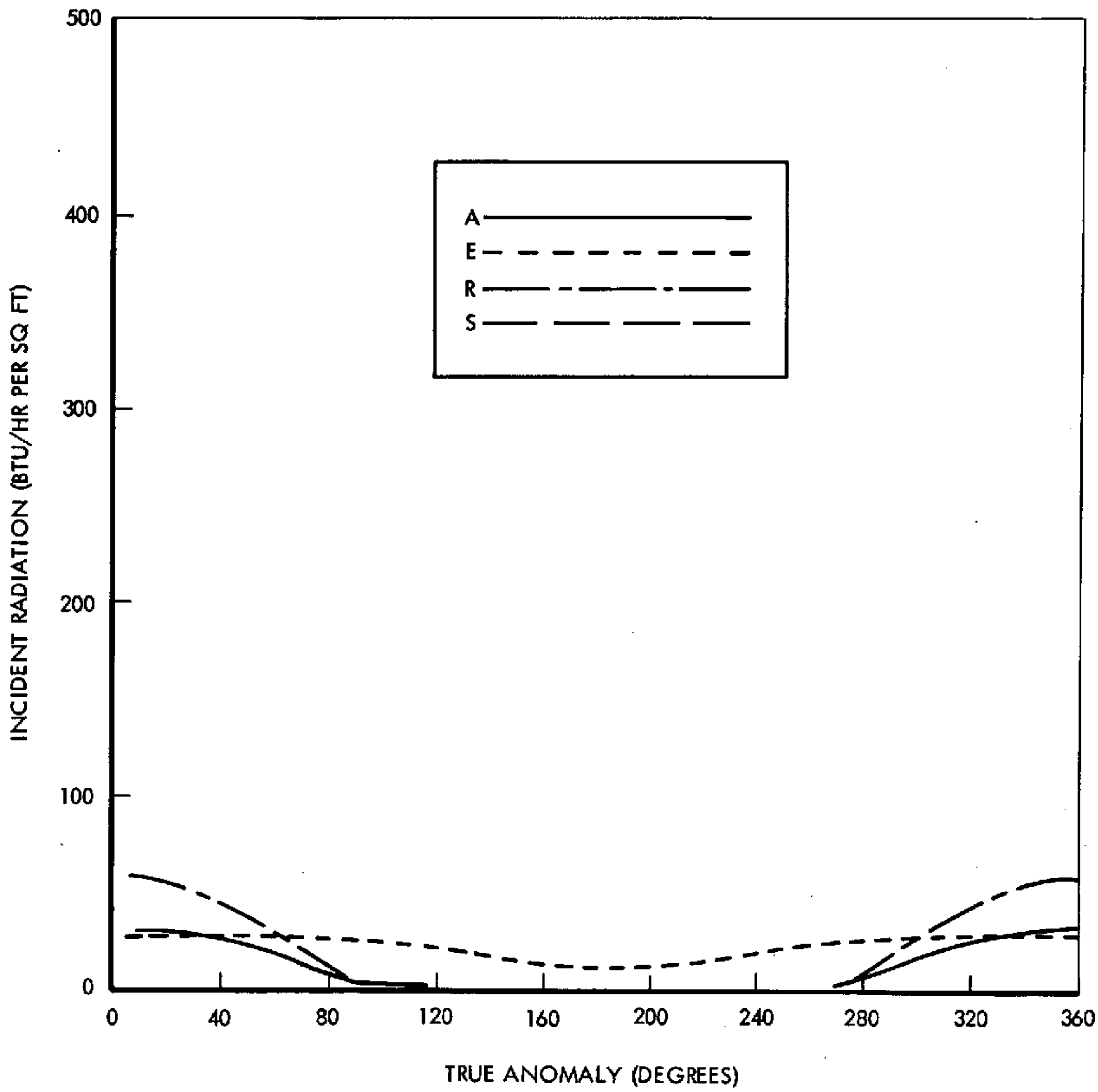


Figure 59. External Heat Load With Module Spin - Side 4

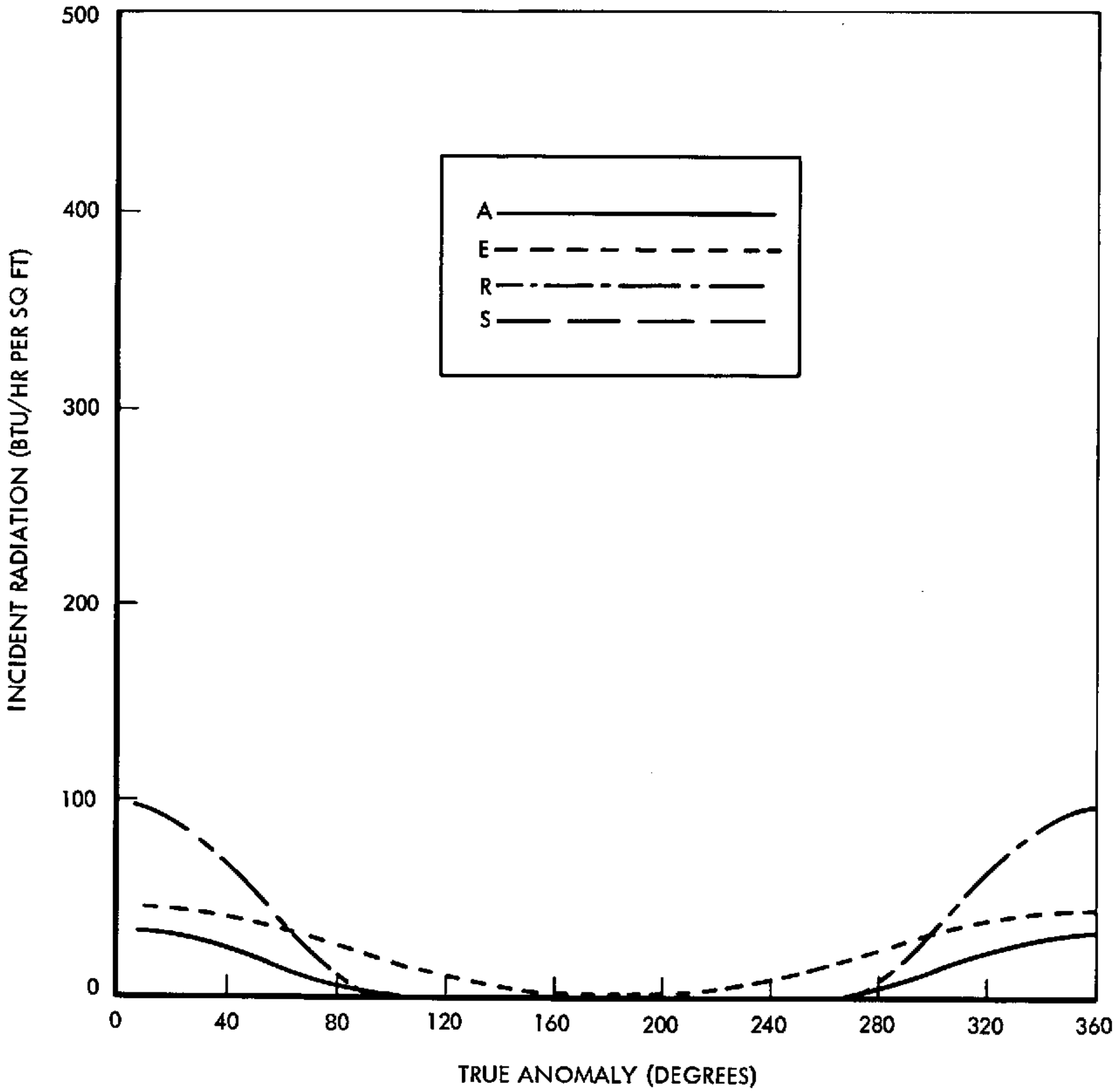


Figure 60. External Heat Load With Module Spin - Side 5

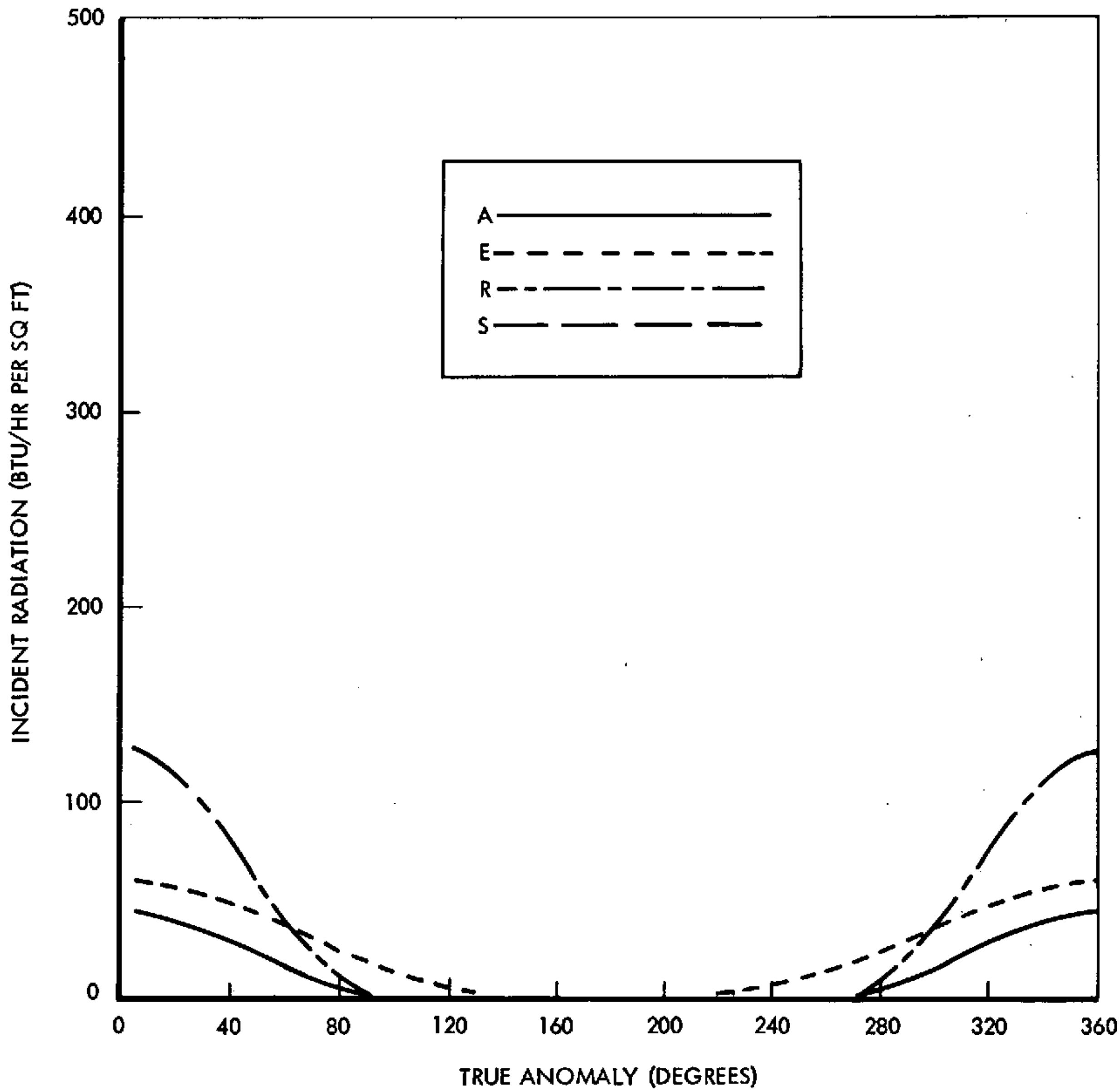


Figure 61. External Heat Load With Module Spin - Side 6

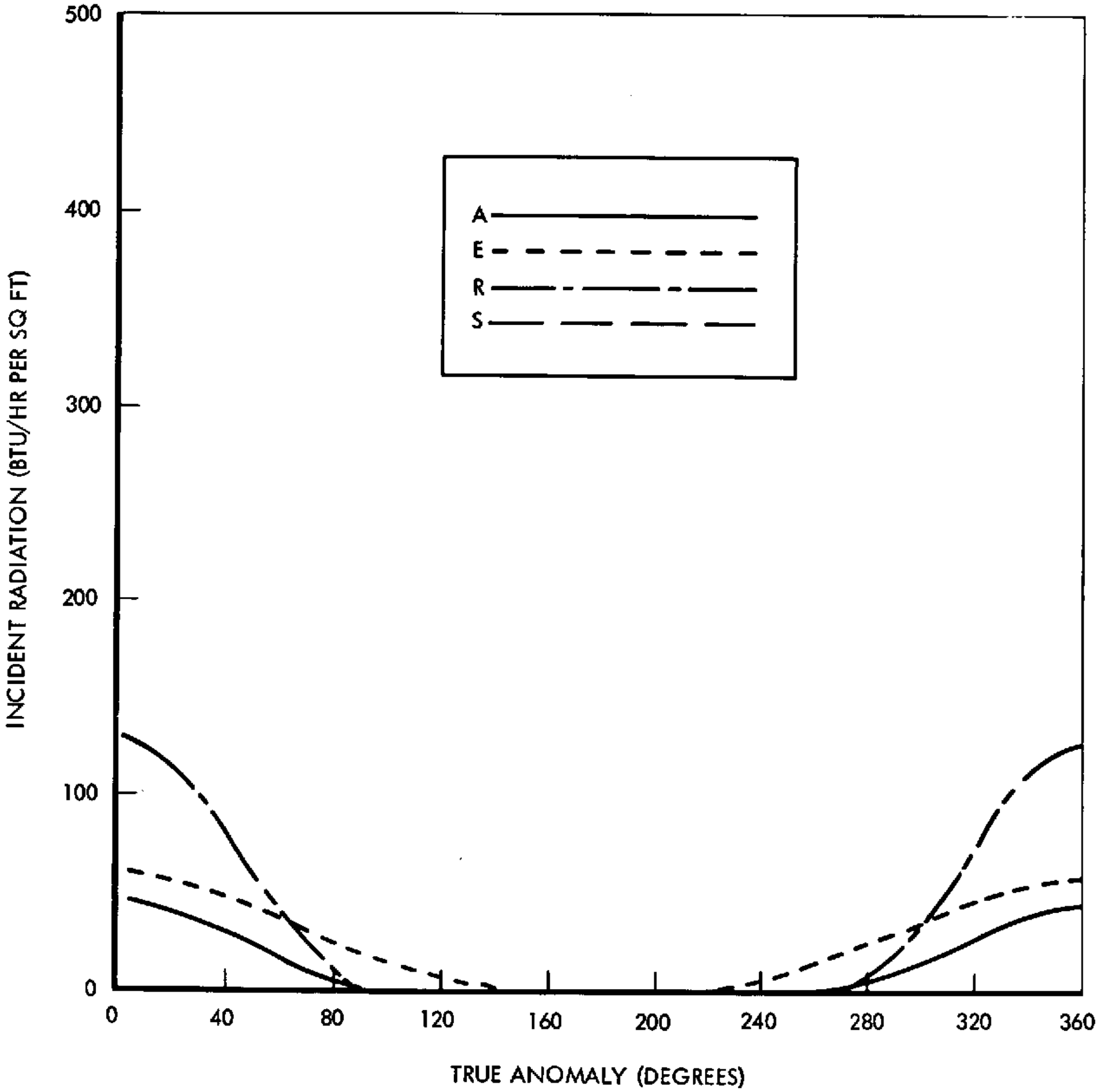


Figure 62. External Heat Load With Module Spin - Side 7

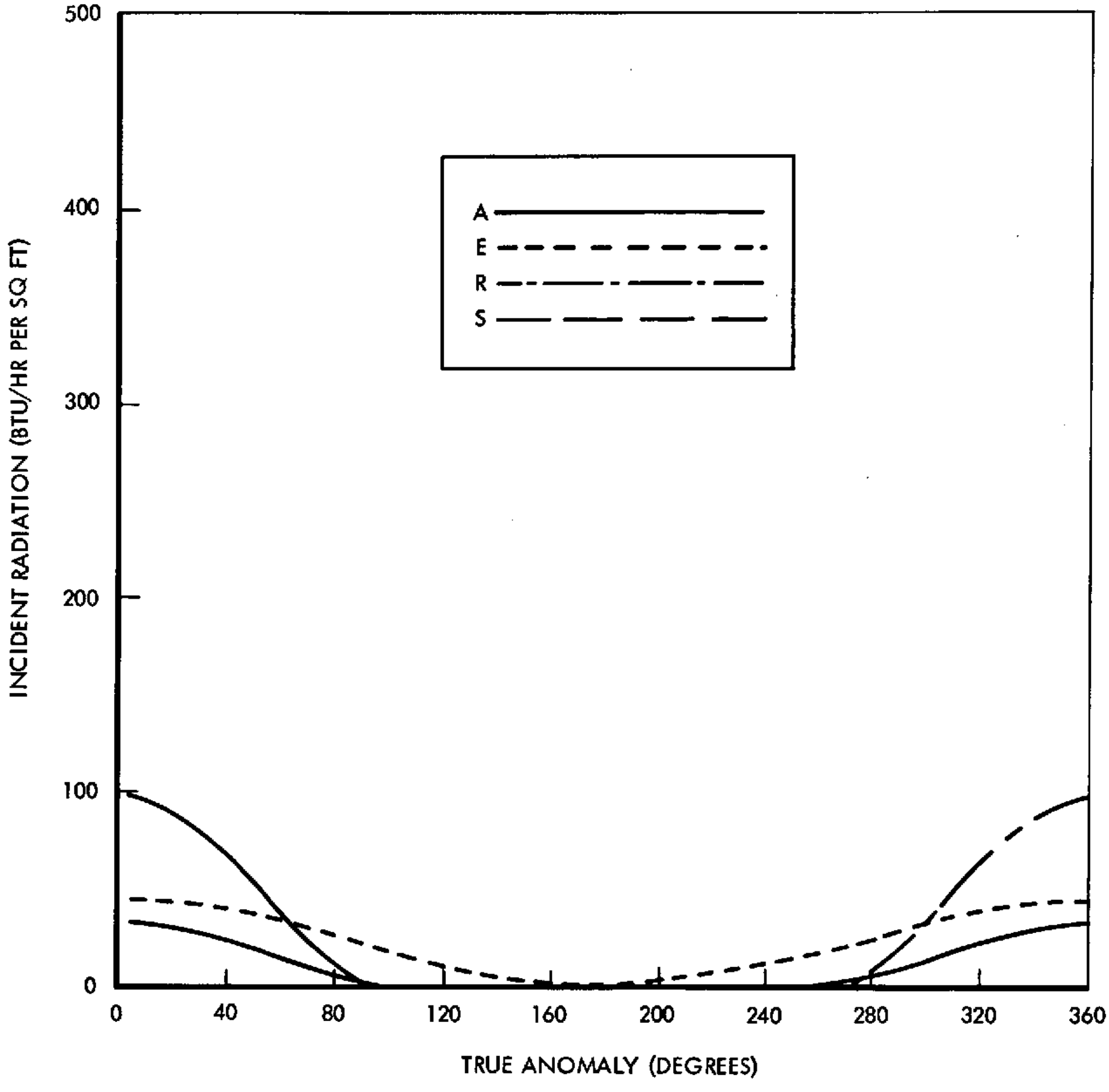


Figure 63. External Heat Load With Module Spin - Side 8

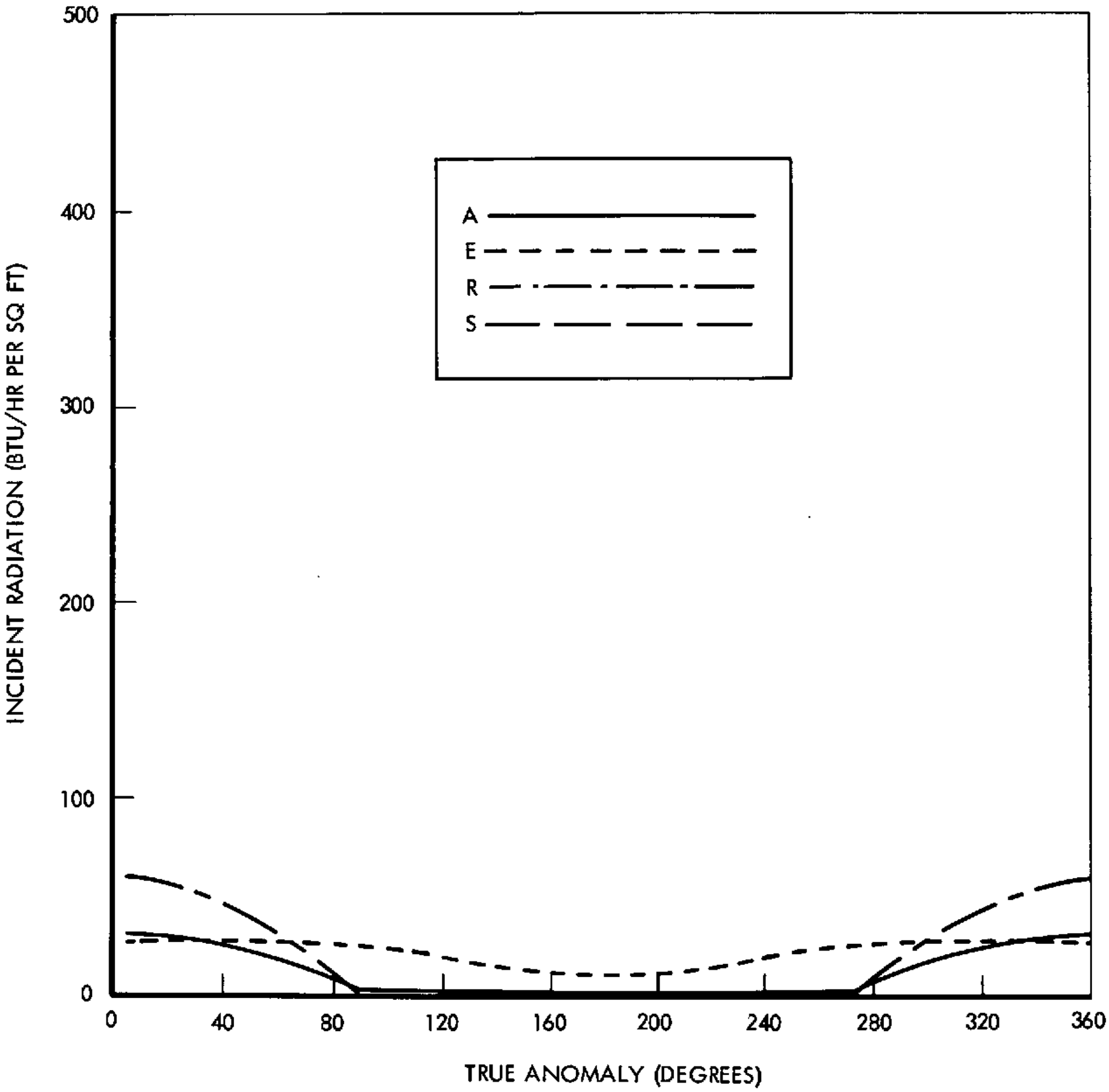


Figure 64. External Heat Load With Module Spin - Side 9

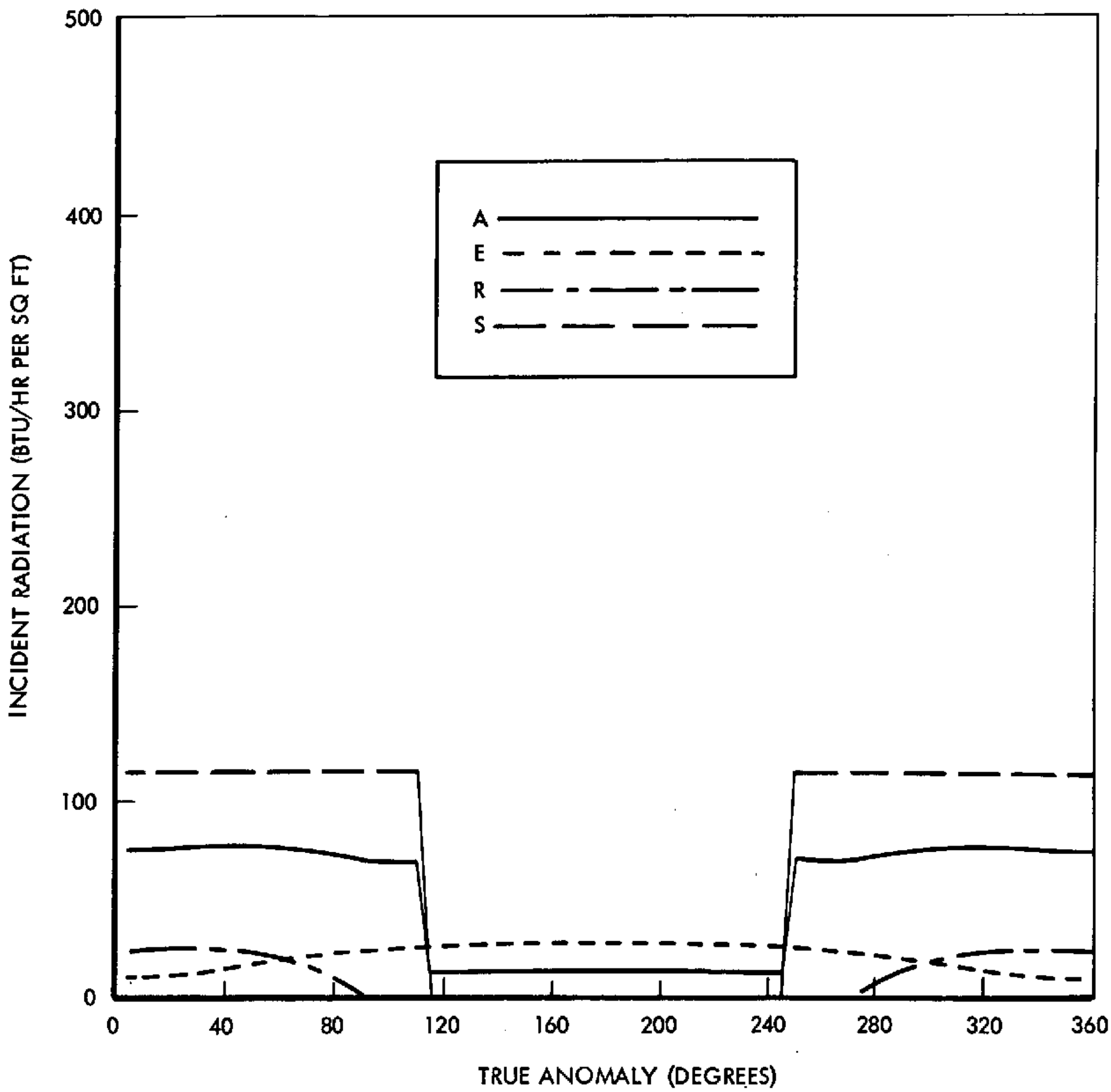


Figure 65. External Heat Load With Module Spin - Side 10

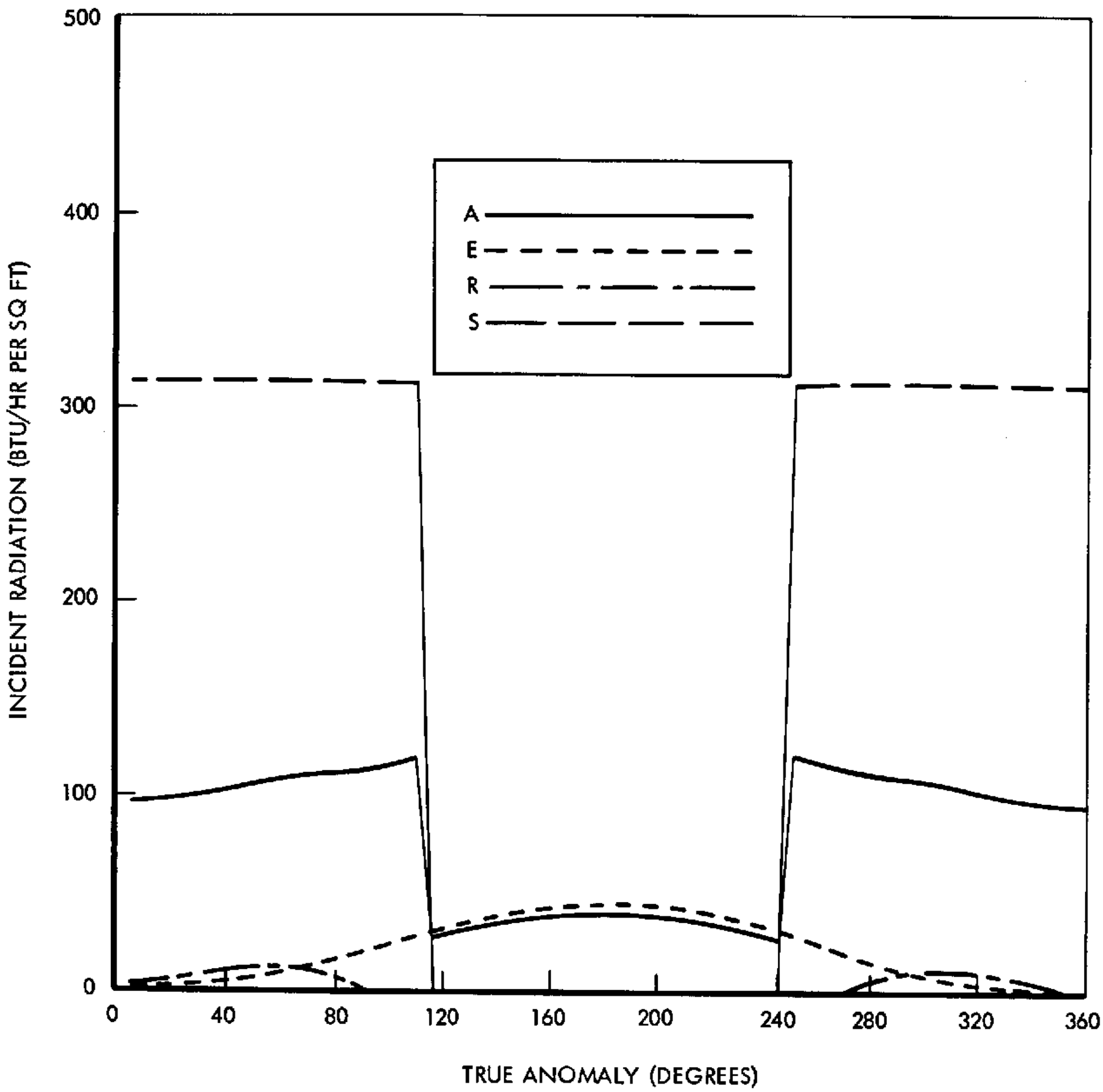


Figure 66. External Heat Load With Module Spin - Side 11

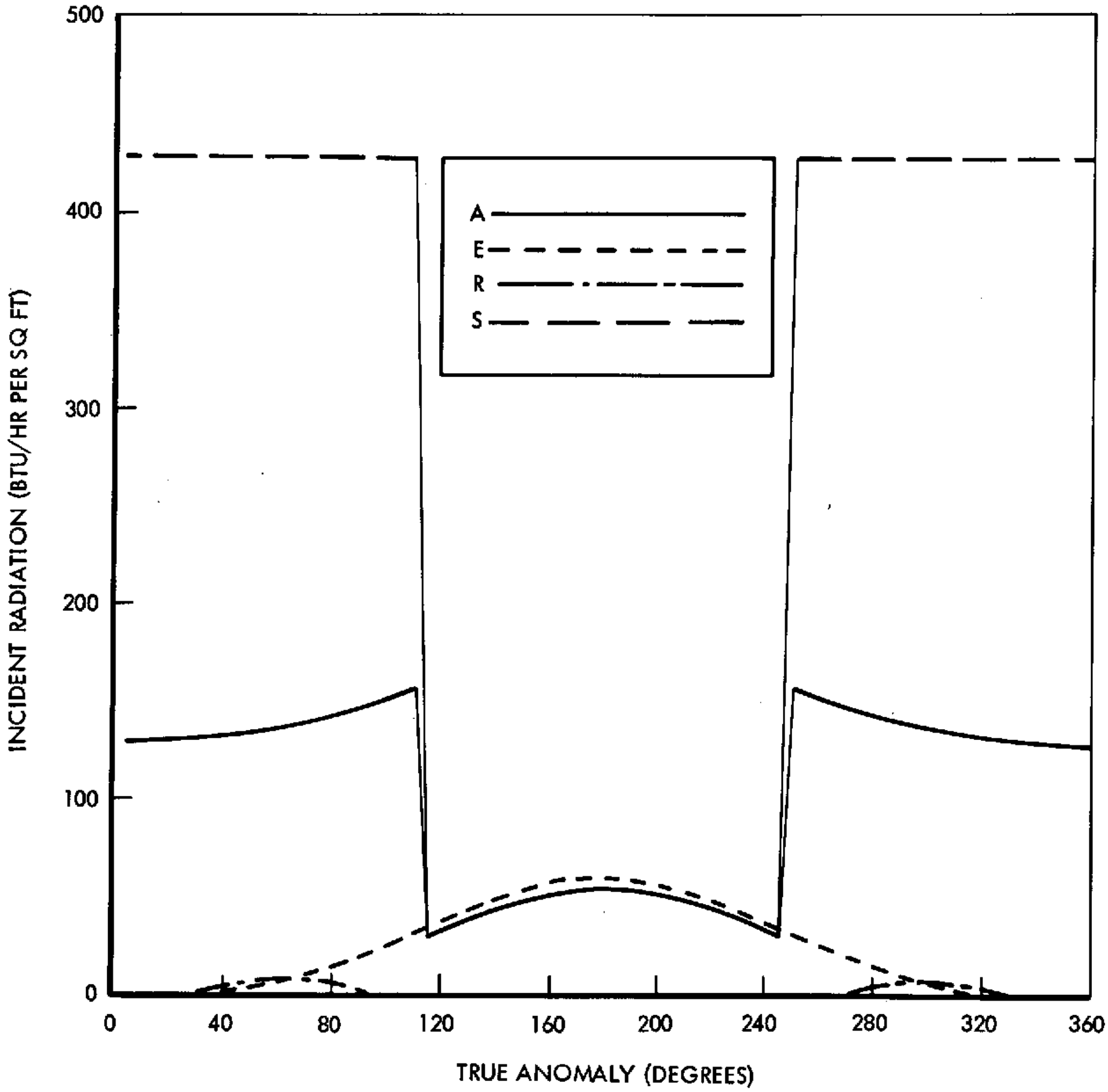


Figure 67. External Heat Load With Module Spin - Side 12

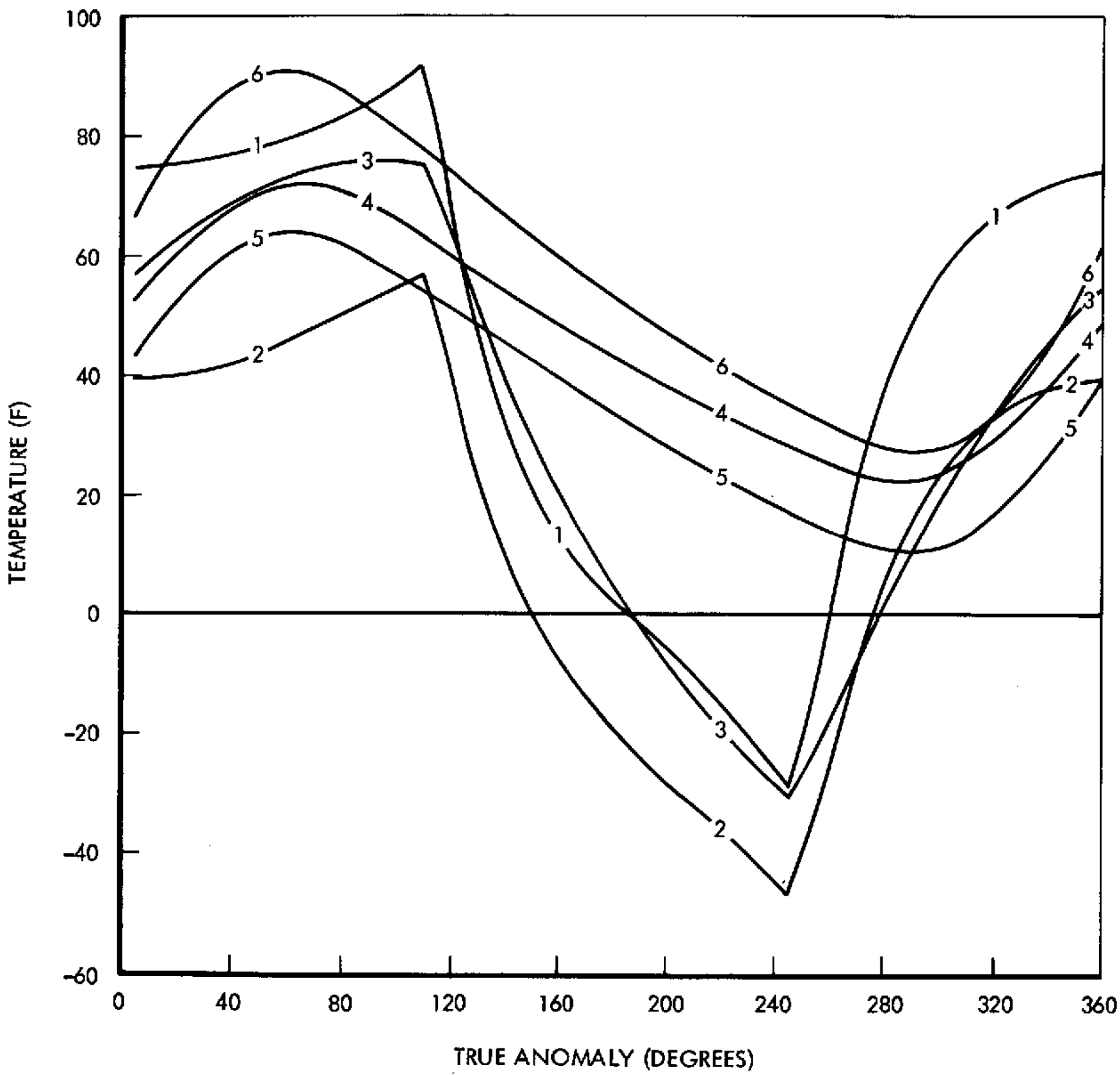


Figure 68. Temperature Profile With Module Spin - Sides 1 to 6

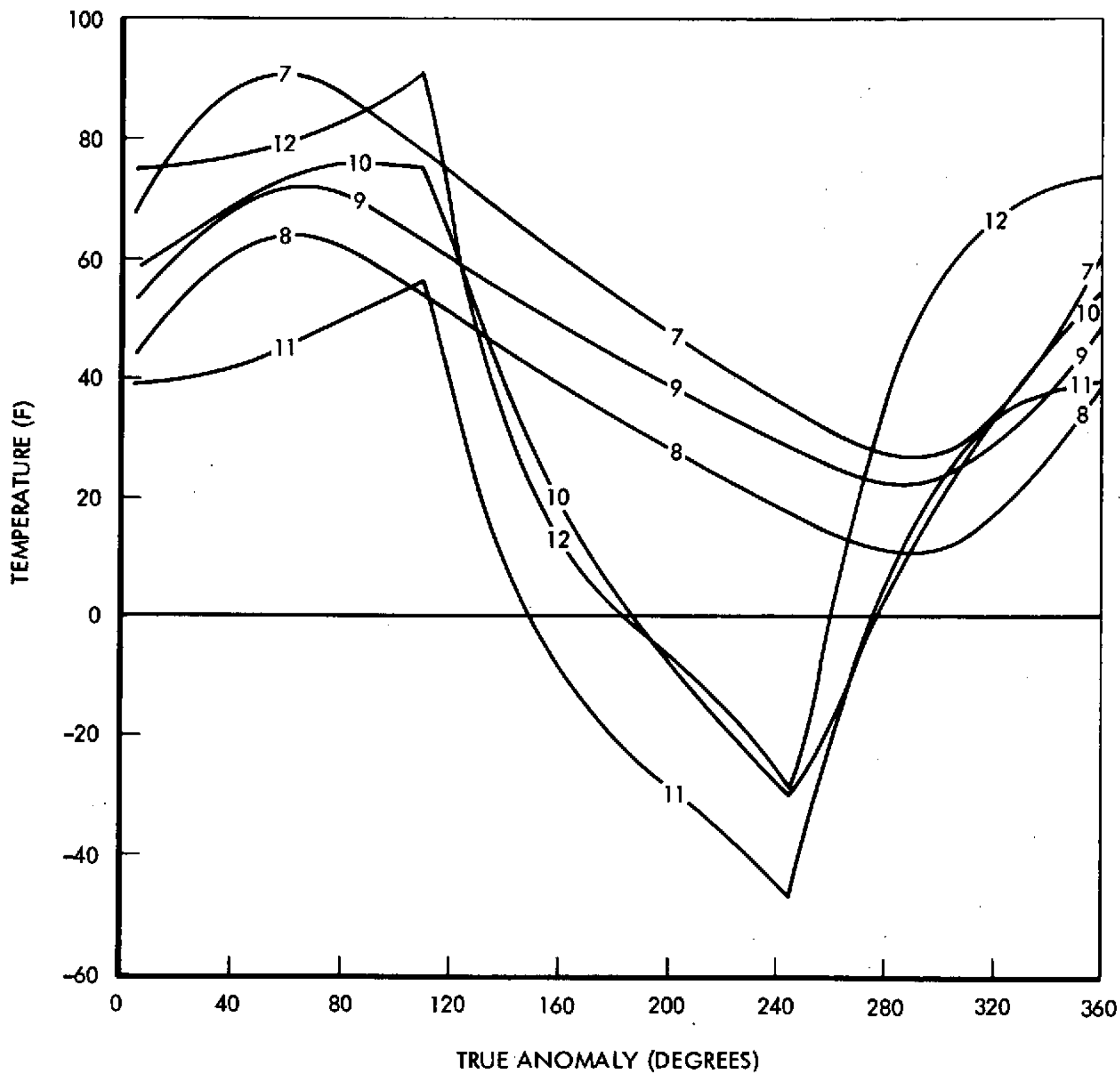


Figure 69. Temperature Profile With Module Spin - Sides 7 to 12

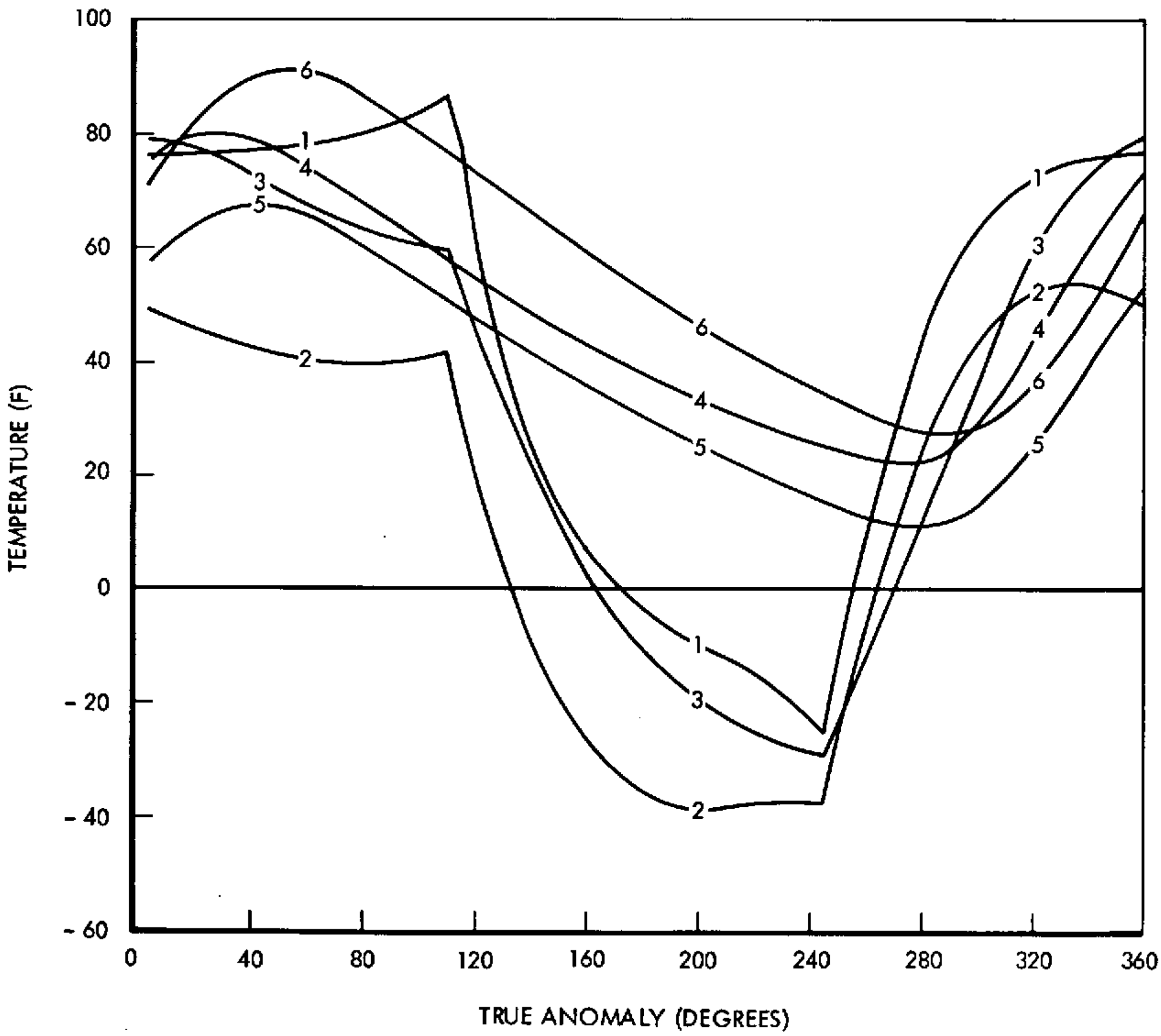


Figure 70. Temperature Profile, No Module Spin - Sides 1 to 6

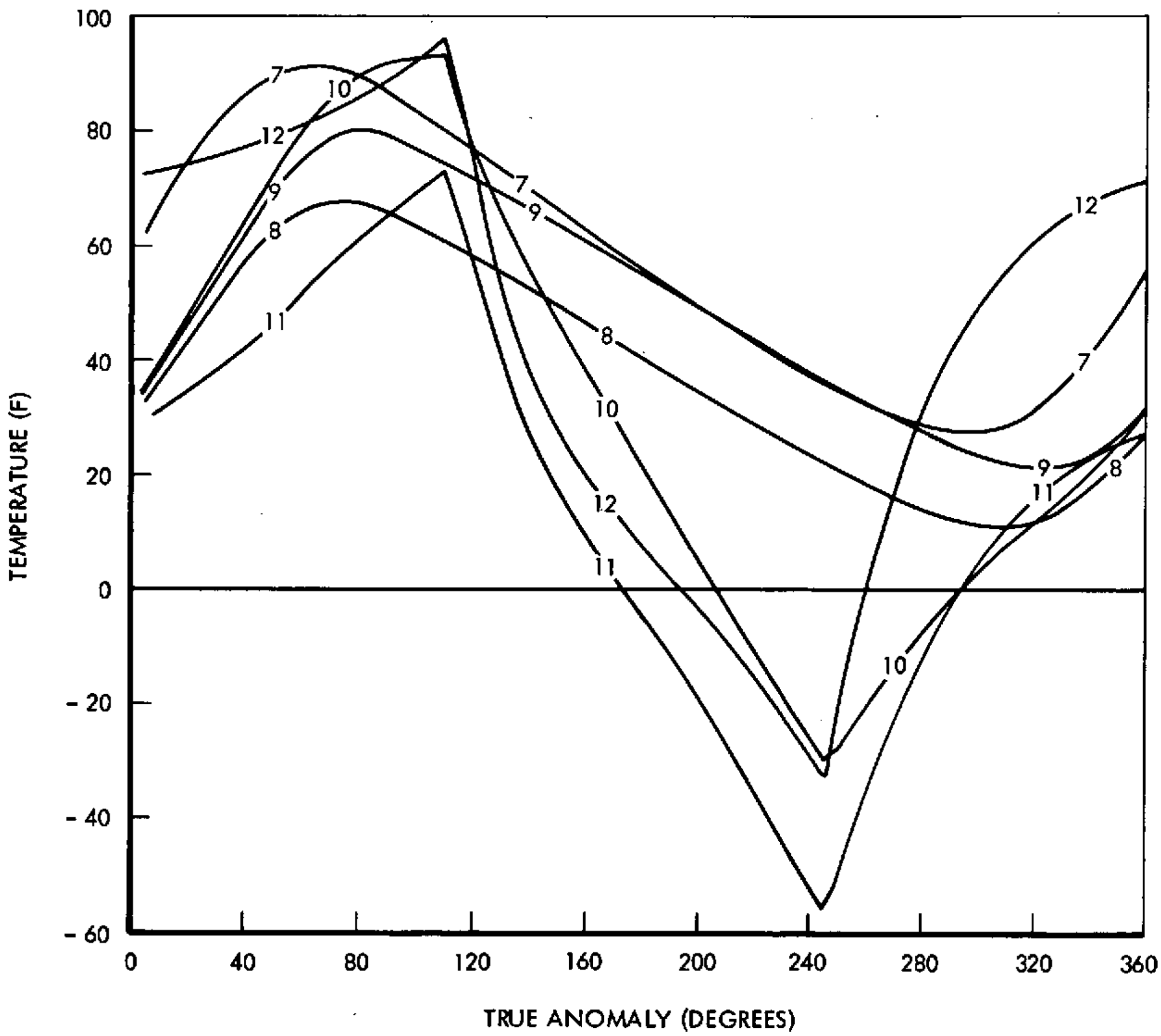


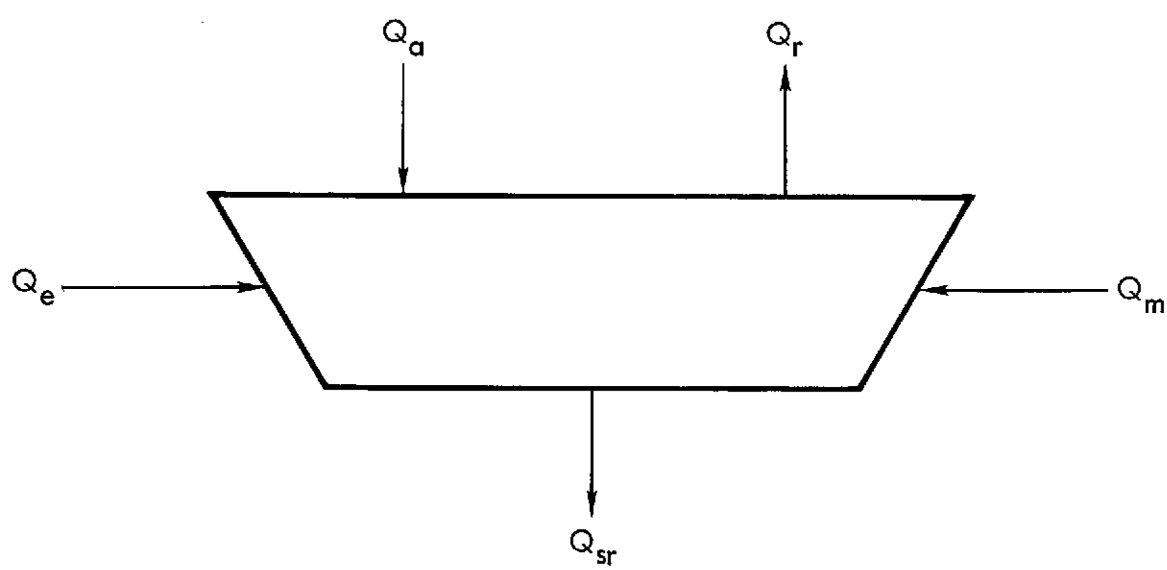
Figure 71. Temperature Profile, No Module Spin - Sides 7 to 12

l Temperature Gradient

Knowing the values of the absorbed radiant heat input, it is possible—by using an electric network analog of the wall—to determine the wall temperature gradient. An IBM computer program is available for this purpose, and in addition gives the radiation heat transfer coefficient from the outside surface to space. Figures 72 through 77 show how the outside and inside module wall temperatures for sides 1 through 6 vary with time in orbit. These temperatures refer to the original wall design using 2 inches of insulation and 1 inch of honeycomb filled with insulation. Figures 78 through 82 show the same temperature profiles for sides 1 through 5, for a wall having one inch of insulation and one inch of honeycomb filled with insulation. The inside wall temperature is more nearly constant in the first case than in the second. Time prevented further investigation of optimum insulation requirements.

Module Heat Balance

The absorbed radiant input curves of Figures 56 through 68 can be graphically integrated to find the total external heat entering the module. The heat radiated to space from the module can be calculated from surface temperatures and radiation heat transfer coefficients, plotted against orbit time, and graphically integrated. A heat balance for the entire module can then be calculated, and the space radiator heat rejection rate found. The method can be illustrated in this manner:



Q_a - Absorbed external radiation

Q_r - Radiation from surface to space

Q_e - Equipment heat load

Q_m - Crew metabolic heat load

Q_{sr} - Heat rejected by space radiator

$$Q_a + Q_e + Q_m = Q_r + Q_{sr}$$

$$Q_{sr} = Q_a + Q_e + Q_m - Q_r = 130,980 + 6800\left(\frac{95.46}{60}\right) + 3000\left(\frac{95.46}{60}\right) - 143,166$$

$$Q_{sr} = 3314 \text{ Btu/orbit}$$

$$= 2075 \text{ Btu/hour}$$

Thus it is seen that the heat duty of the space radiator is 2075 Btu/hour.

It should be noted that different values of the space radiator heat duty will be found if it is assumed that heat losses through the module wall are zero. From the above analysis—using the values of surface absorptivity and emissivity listed above—it is seen that the heat radiated from the surface is greater than the external heat absorbed. Thus, the space radiator is not required to reject all of the heat generated on the inside.

Heat Rejection System

Heat may be rejected from a space vehicle either by active or passive systems. In the active system, cabin air is passed through a heat exchanger cooled by a vapor compression-type refrigerator. Heat is removed by condensing the refrigerant in a condenser from which the heat is rejected by a liquid flowing through a space radiator. Use of the active system is necessary when the surface of the radiator has an emissivity to absorptivity ratio less than 6. The increased power penalty of the refrigerant system, plus the added equipment, makes the active system less desirable than the passive. The passive system used an air-liquid heat exchanger and a space radiator to reject the heat transferred from the air to the liquid. It is a simpler system with fewer moving parts. Assuming that, by 1965, materials with satisfactory surface characteristics for radiators will be available, it was decided to use a passive heat rejection system. (The operation of the space radiator-glycol loop is discussed later.)

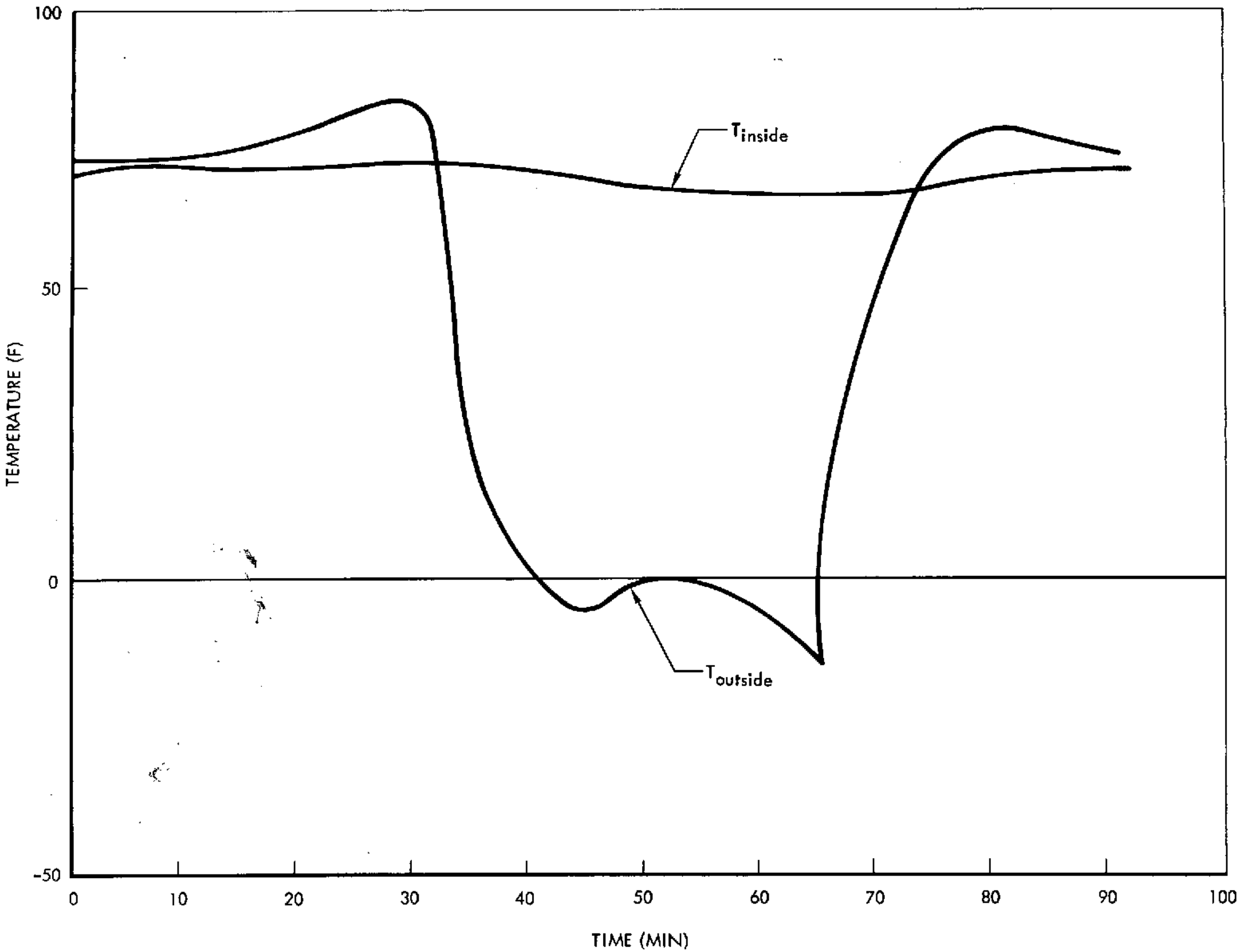


Figure 72. Module Wall Temperatures Versus Orbit Time With 2-Inch Insulation - Side 1

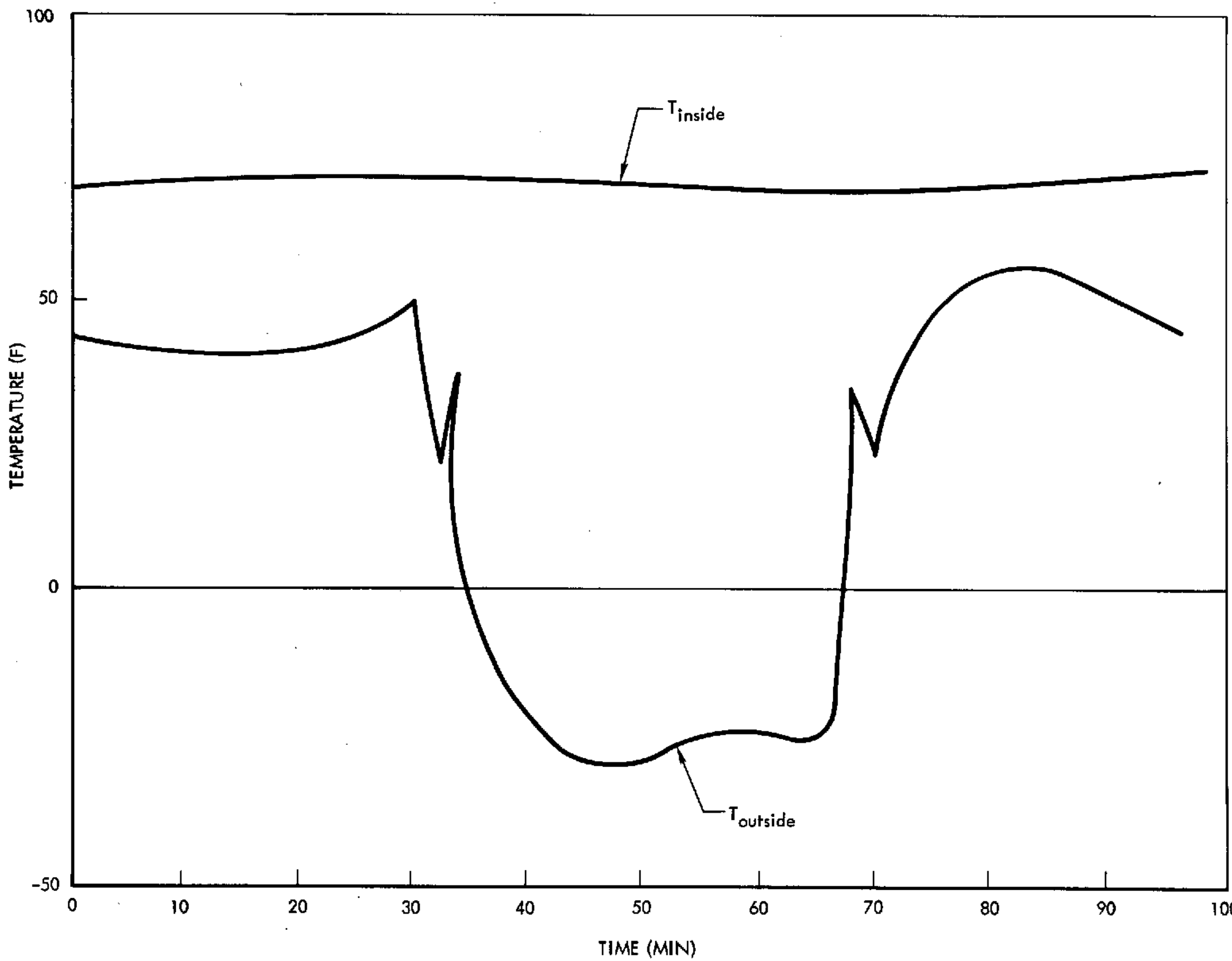


Figure 73. Module Wall Temperatures Versus Orbit Time With 2-Inch Insulation - Side 2

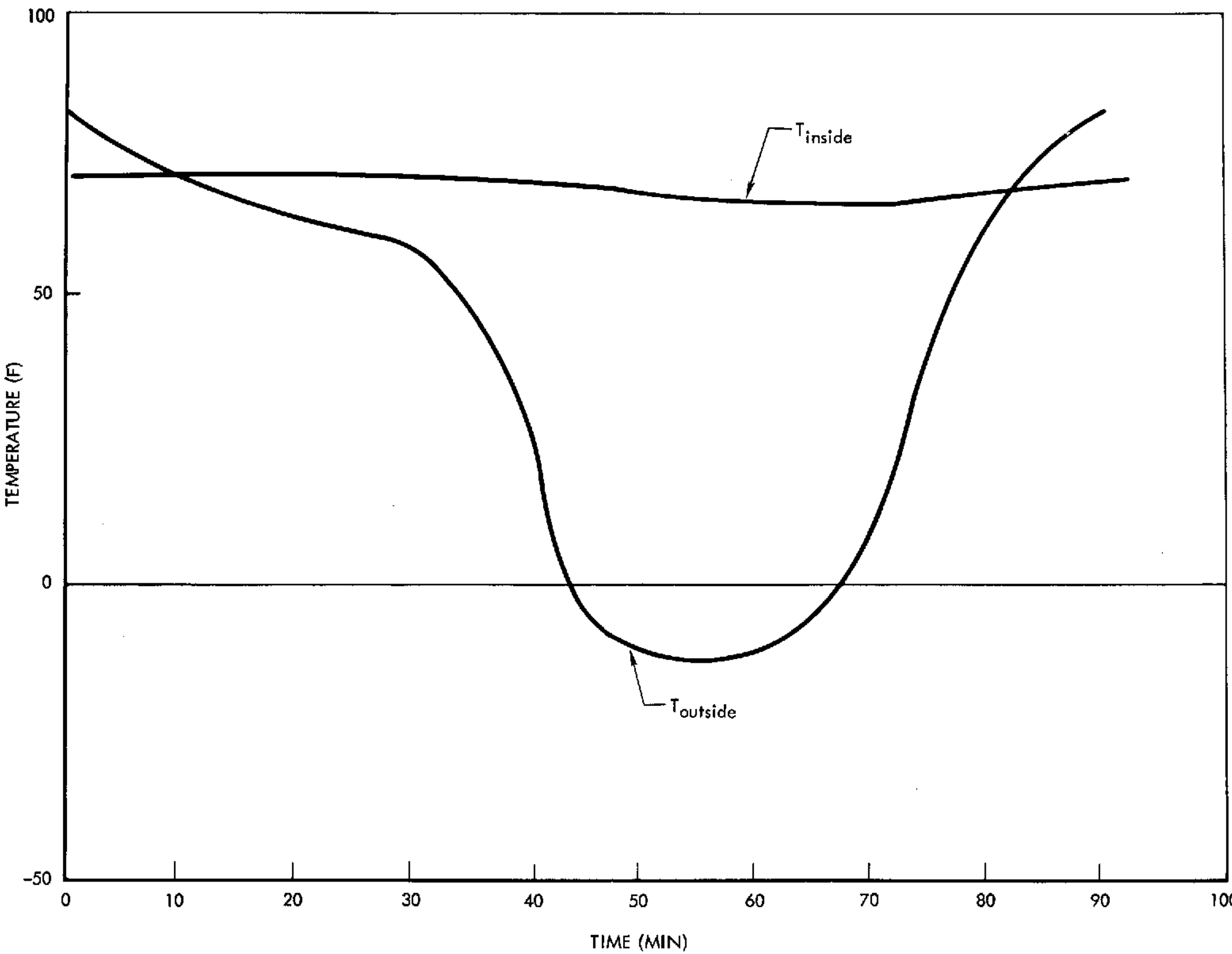


Figure 74. Module Wall Temperatures Versus Orbit Time With 2-Inch Insulation - Side 3

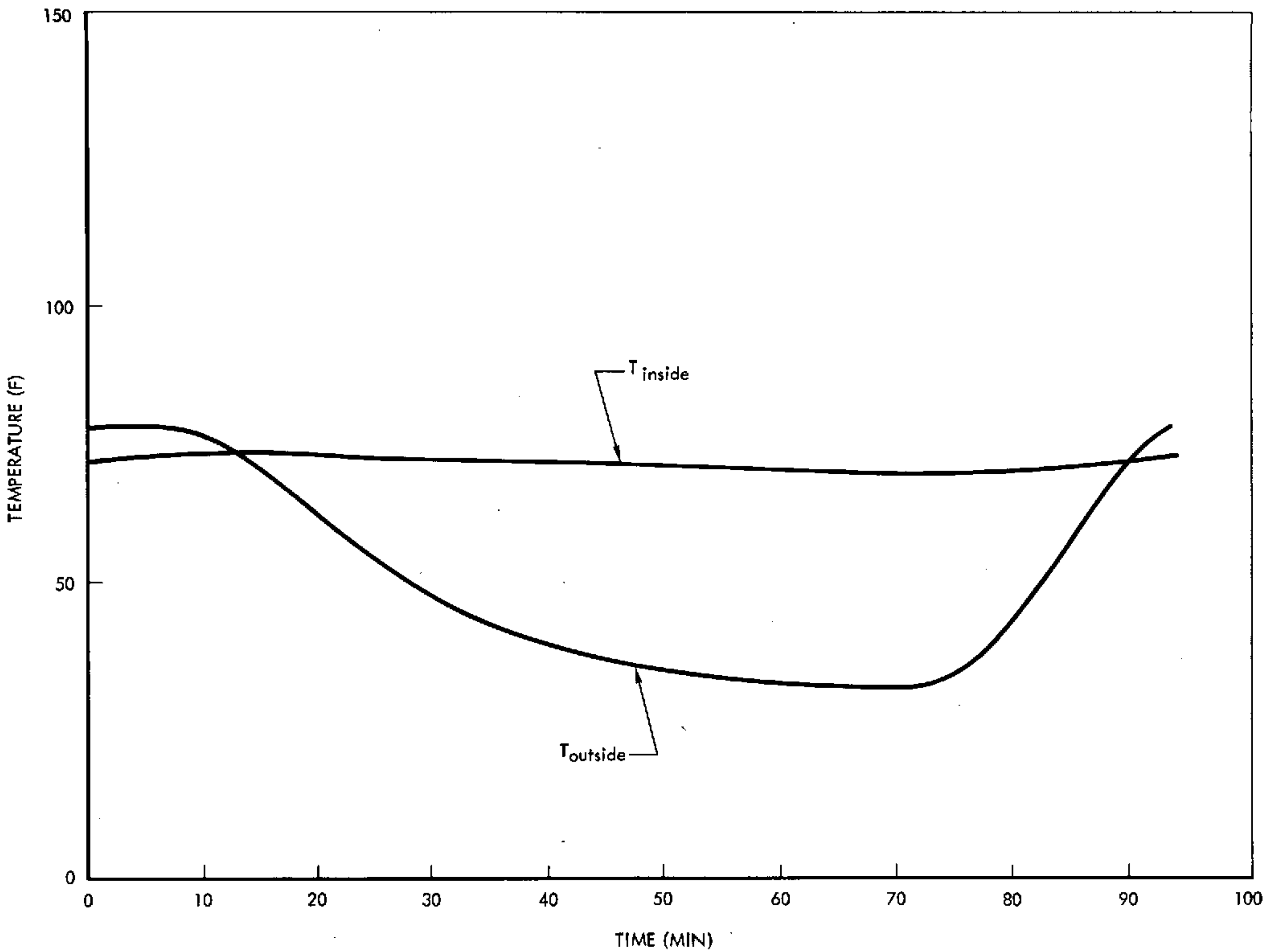


Figure 75. Module Wall Temperatures Versus Orbit Time With 2-Inch Insulation - Side 4

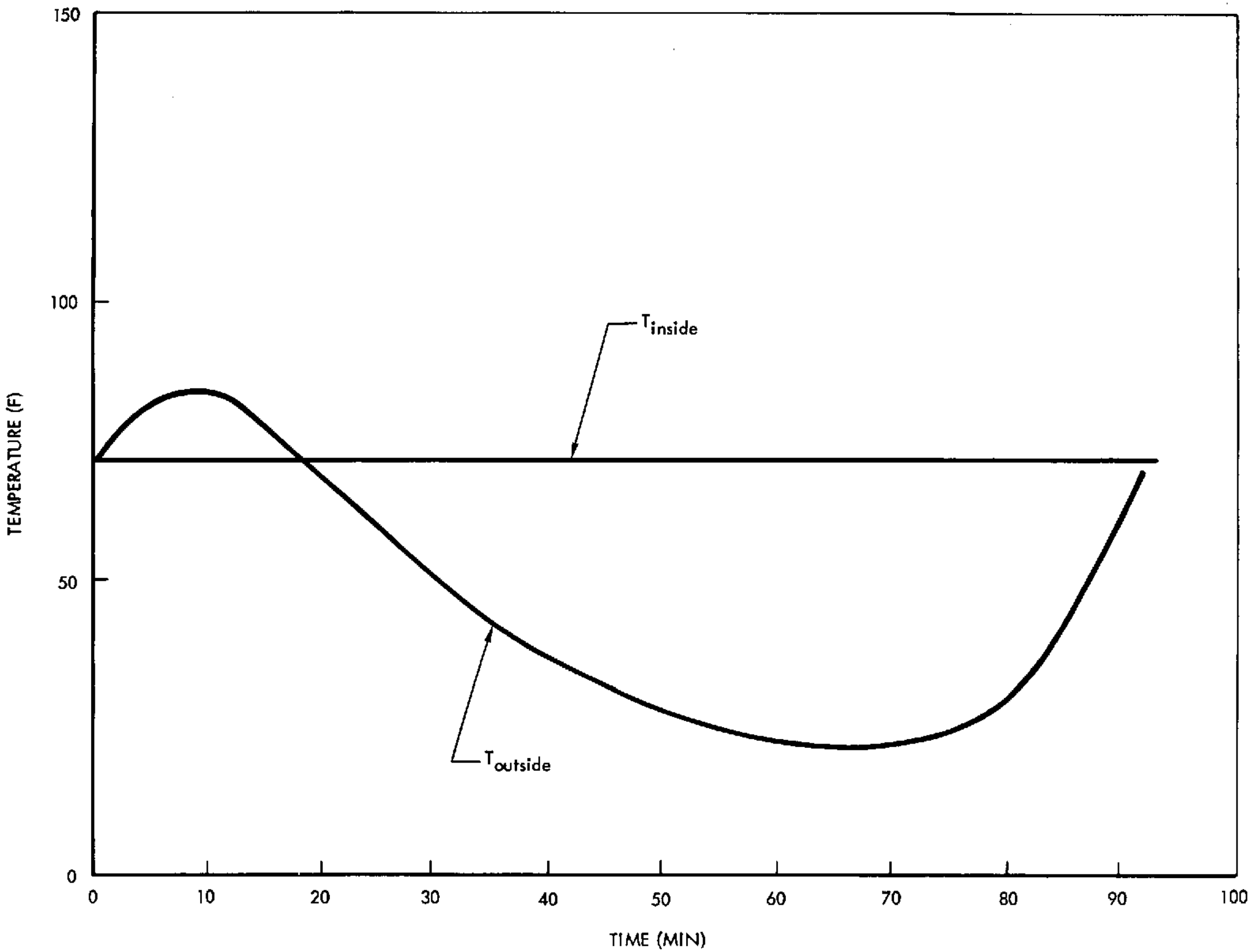


Figure 76. Module Wall Temperatures Versus Orbit Time With 2-Inch Insulation - Side 5

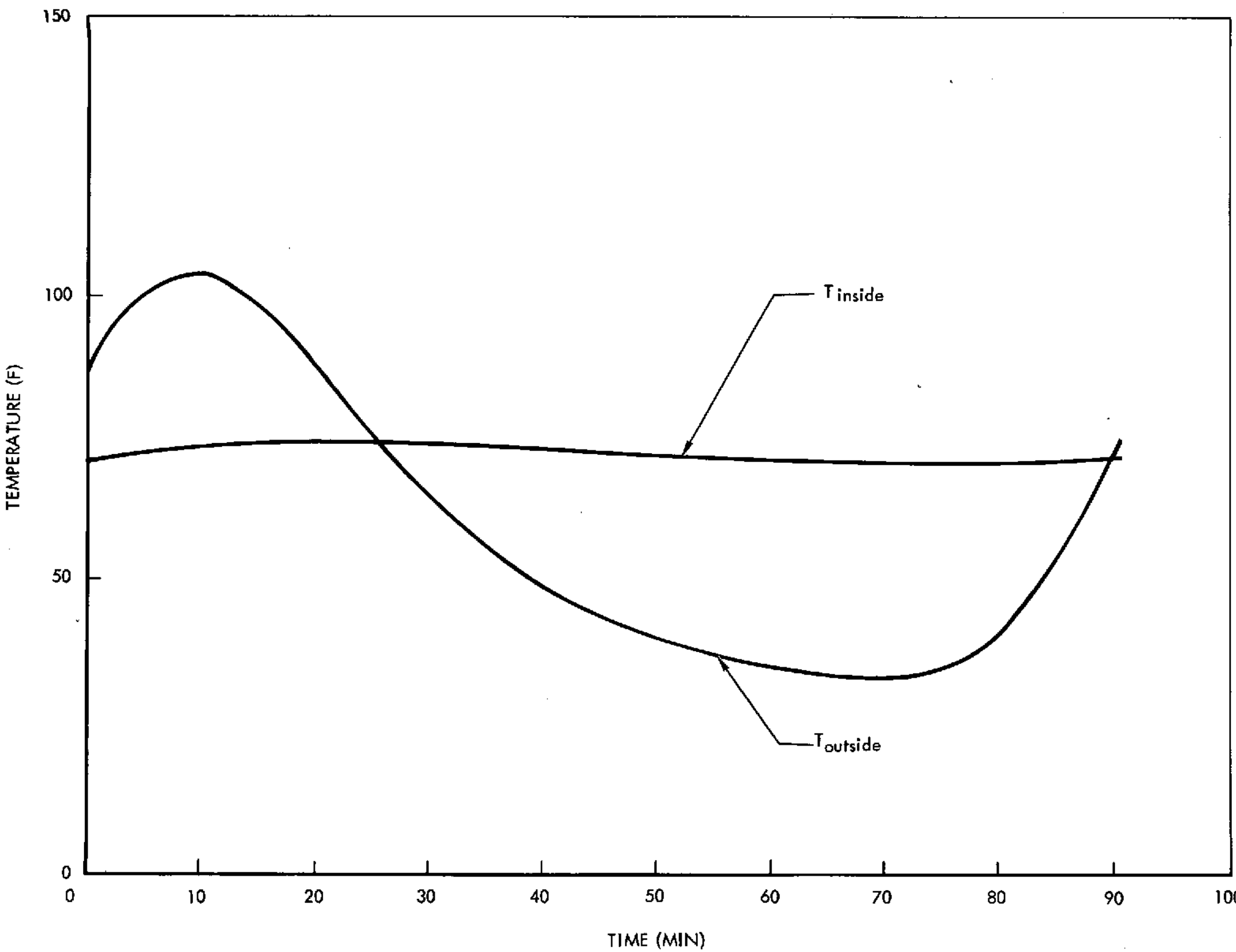


Figure 77. Module Wall Temperatures Versus Orbit Time With 2-Inch Insulation - Side 6

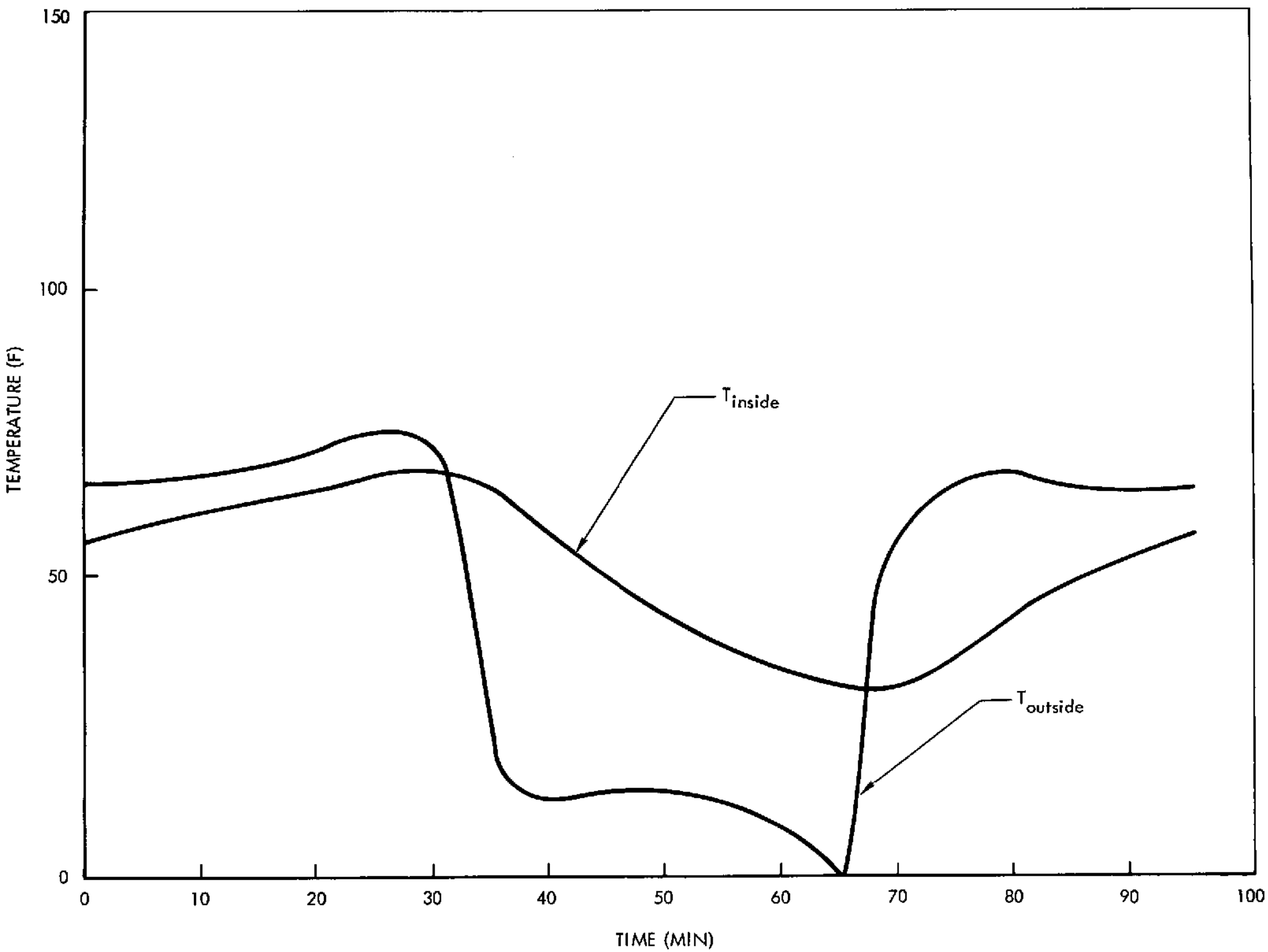


Figure 78. Module Wall Temperatures Versus Orbit Time With 1-Inch Insulation - Side 1

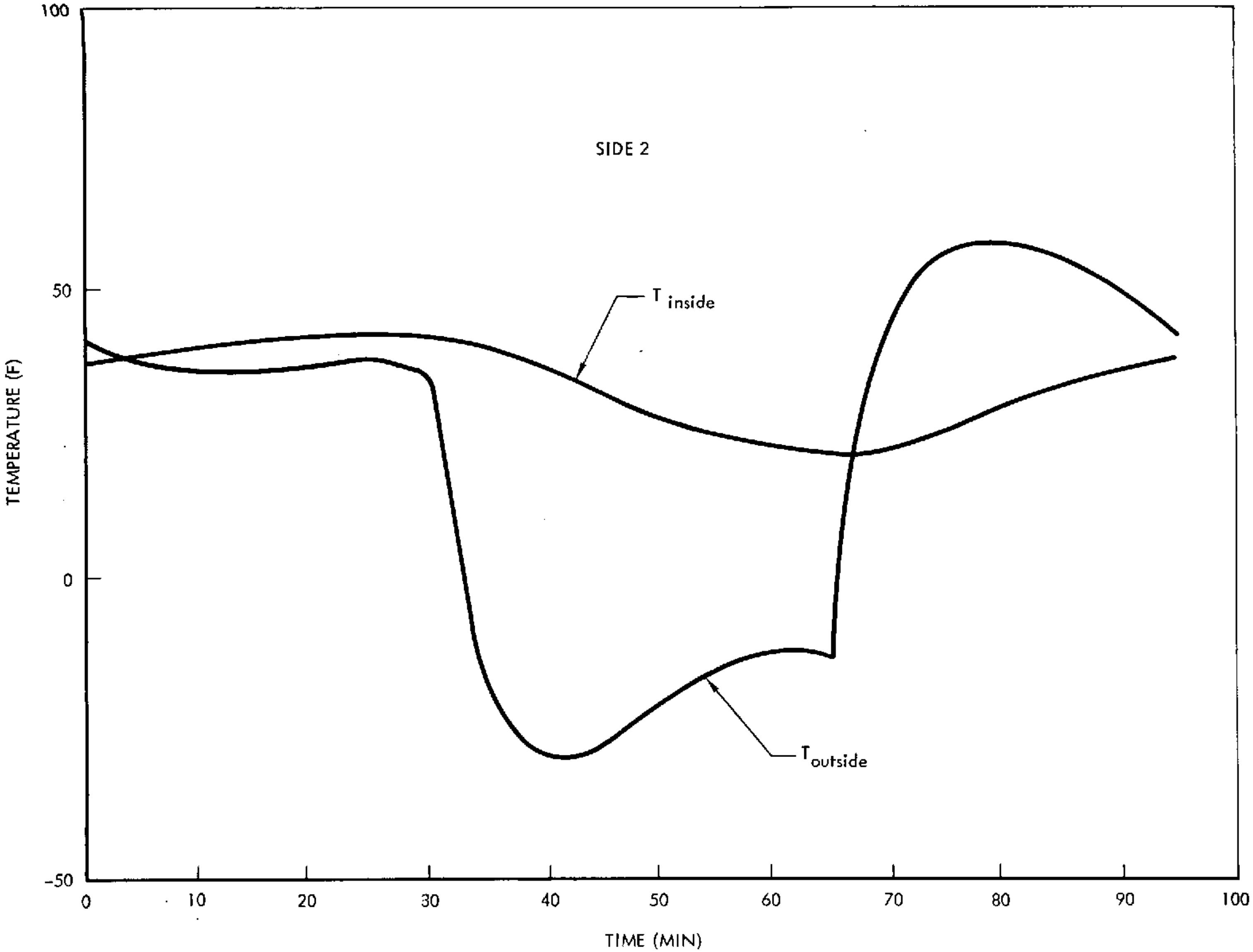


Figure 79. Module Wall Temperatures Versus Orbit Time With 1-Inch Insulation - Side 2

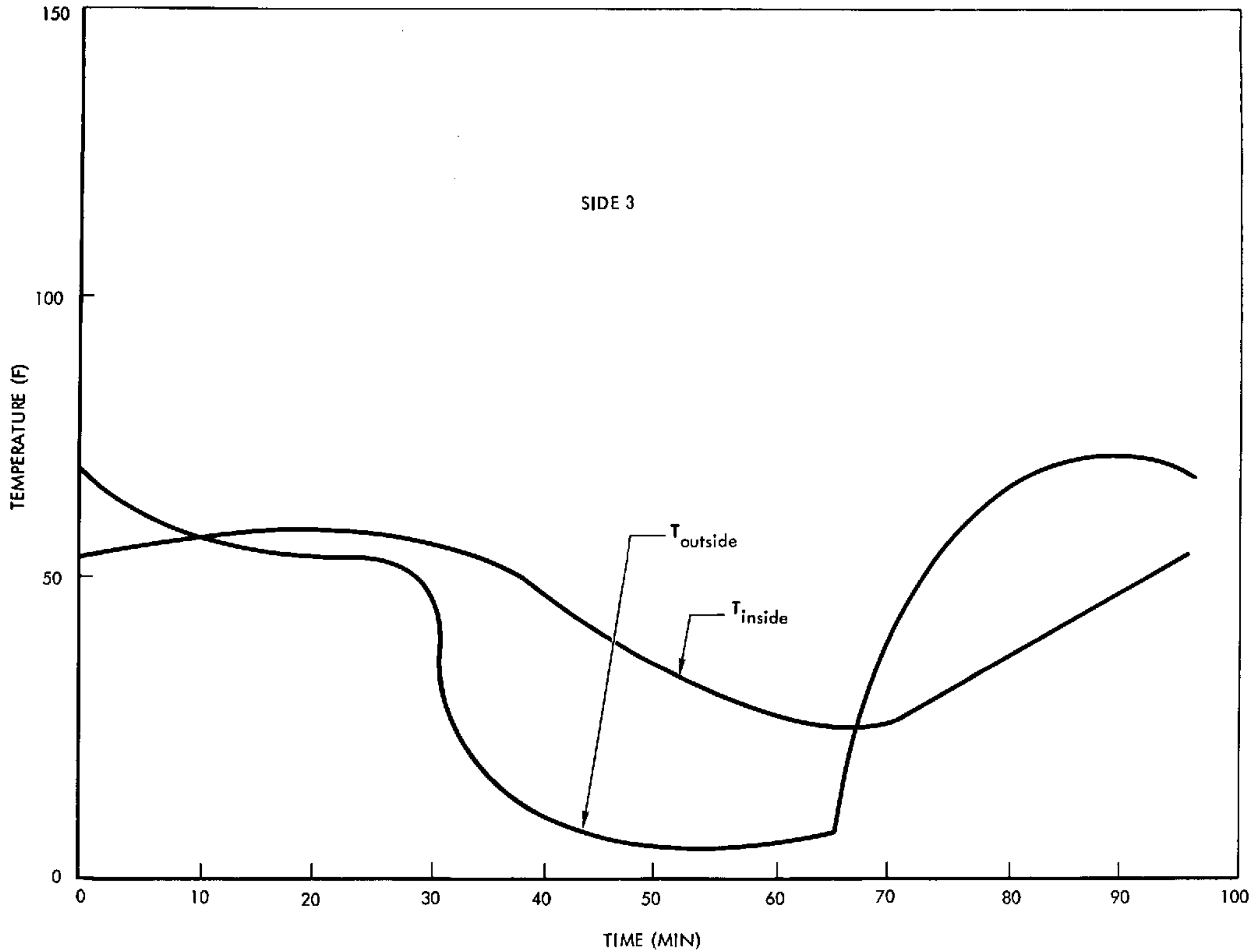


Figure 80. Module Wall Temperatures Versus Orbit Time With 1-Inch Insulation - Side 3

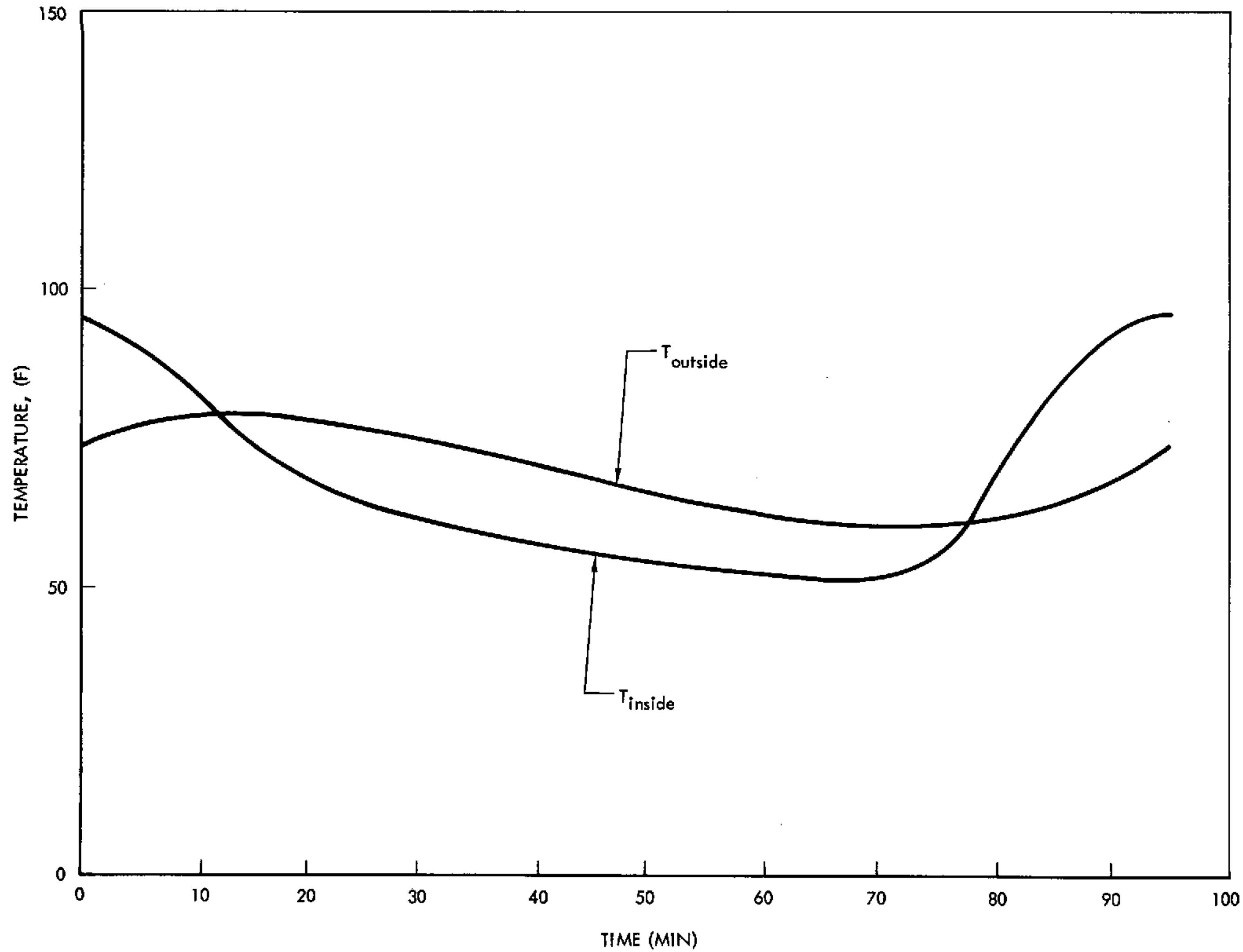


Figure 81. Module Wall Temperatures Versus Orbit Time With 1-Inch Insulation - Side 4

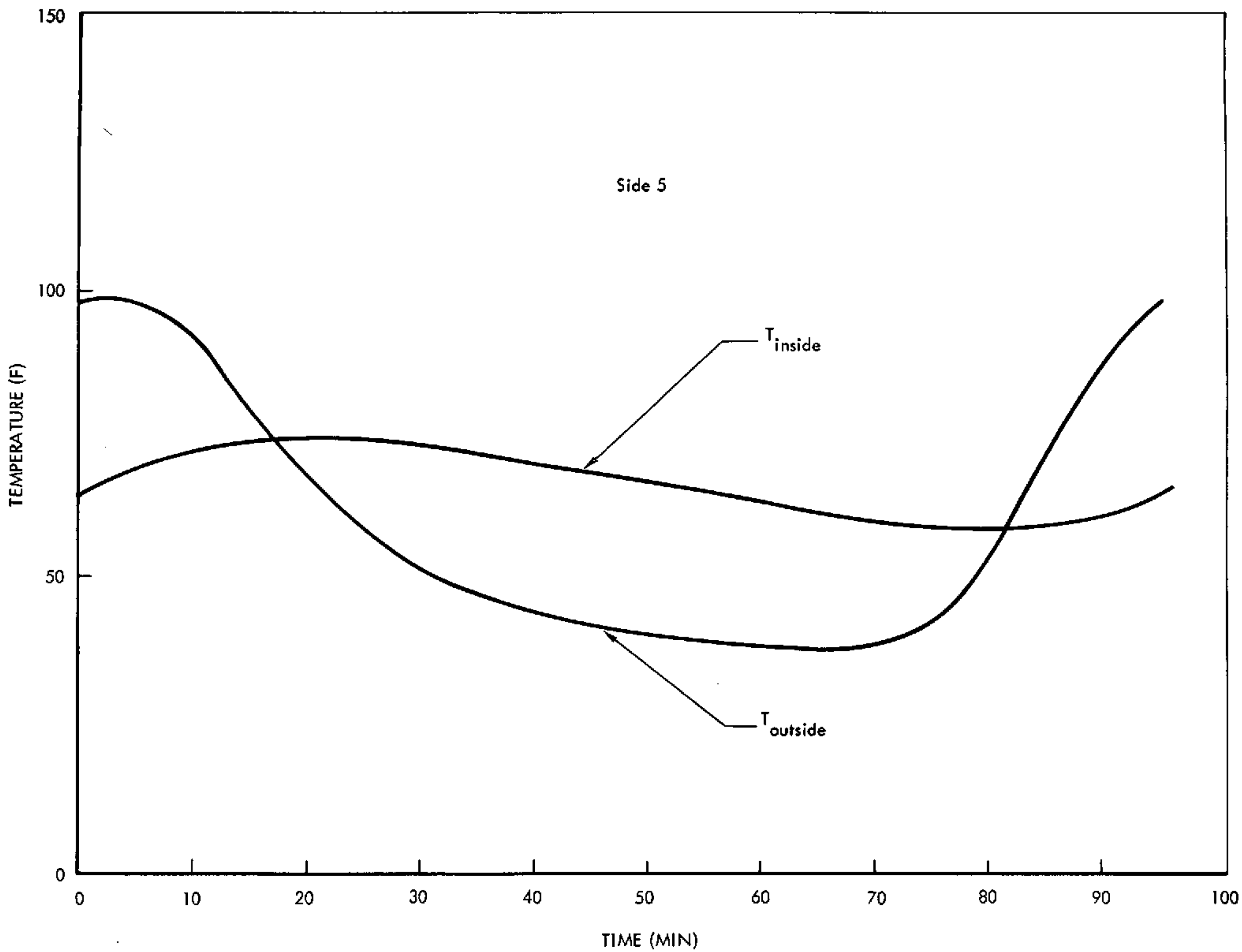


Figure 82. Module Wall Temperatures Versus Orbit Time With 1-Inch Insulation - Side 5

ATMOSPHERIC CONTROL SUBSYSTEM

The principal functions of the atmospheric control subsystem are

1. To circulate and distribute the air within the module
2. To control humidity
3. To supply and control oxygen
4. To supply and control nitrogen
5. To control total module pressure
6. To remove and control carbon dioxide
7. To control contaminants and odors

This subsystem, in conjunction with the thermal control subsystem, provides the means for controlling temperature and rejecting heat for the overall heat balance. All functions listed above are integrated into a piping system containing the necessary equipment for carrying out the necessary operations.

System Development

Because of the extended mission duration of the vehicle, it seemed obvious that a regenerative atmospheric control system was necessary. The weight of expendable materials and stored supplies becomes prohibitive for a vehicle of the size projected here. Although it is intended to resupply materials to the space laboratory periodically, the initial system investigated was of the regenerative type. This system is shown in Figure 83.

As a matter of interest, calculations were made to determine daily makeup requirements of oxygen, nitrogen, water, and lithium hydroxide for five cases: (1) completely open, once through air circulation; (2) carbon dioxide removal by lithium hydroxide, condensation of humidity water, and oxygen and nitrogen makeup with air circulation; (3) same as case 2, but carbon dioxide removed by molecular sieves; (4) same as case 3, but with additional water recovery from urine; (5) same as case 4, with additional recovery of oxygen from carbon dioxide. Comparison of the five cases showed what had been expected—case 5 was the best for extended missions.

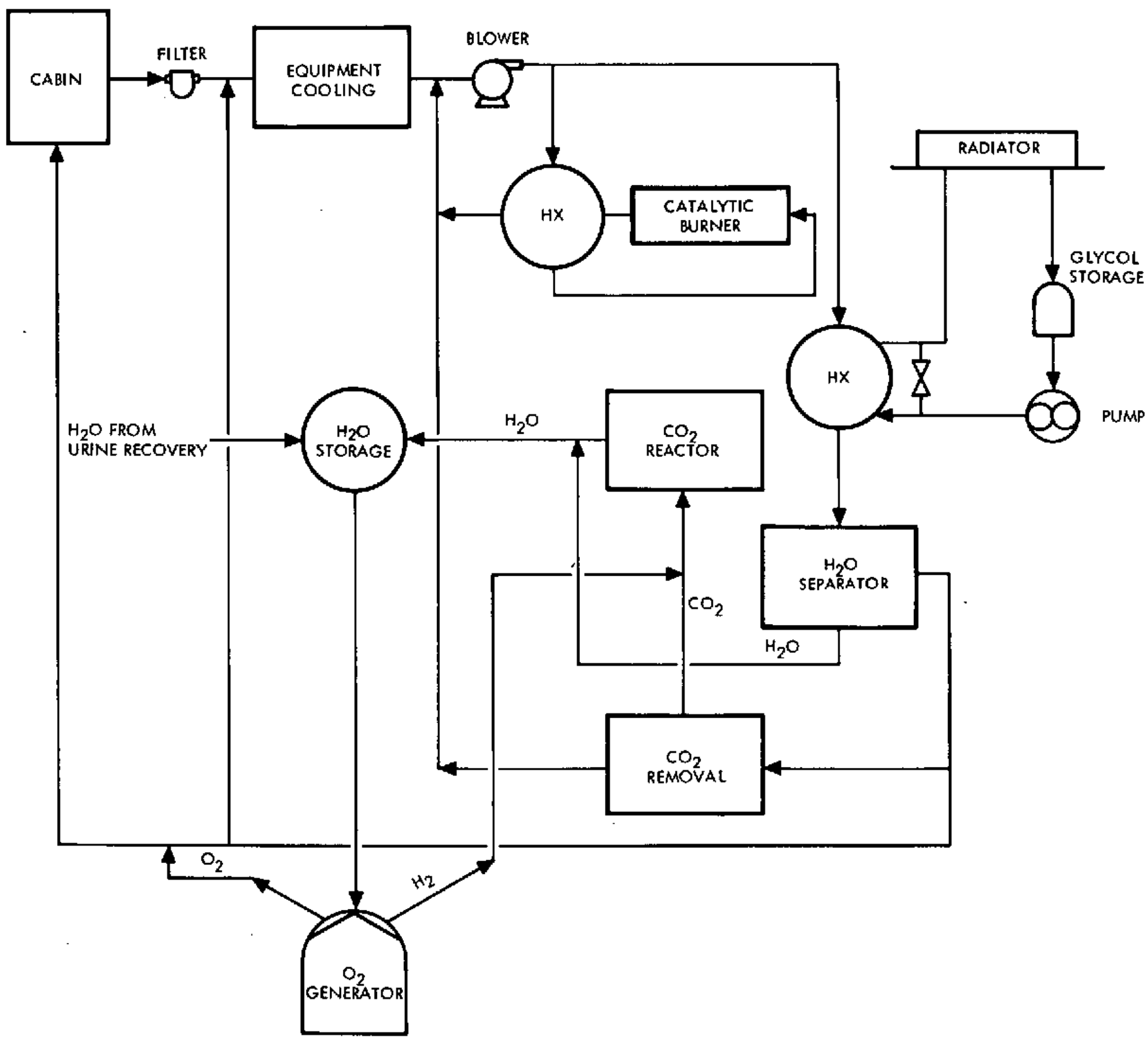


Figure 83. Regenerative Atmospheric Control System Schematic

System Operation

Referring to Figure 83, the atmospheric control system is operated as follows: (1) cabin air is drawn through the filter for removal of dust and other particulate matter, then through the electronic equipment package for cooling; (2) the air then enters the main blower, from which it is sent to the air-glycol heat exchange and the catalytic burner.

In the air-glycol heat exchanger, the air is cooled to its dewpoint by circulating glycol. From the exchanger, the air—now containing droplets of entrained water—enters the centrifugal water separator which removes and coalesces the entrained water and sends it to the water storage tank.

Approximately 1 percent of the air leaving the blower passes through a regenerative heat exchanger and then into the catalytic burner, for control of odors and contaminants; subsequently the air returns to the cabin.

From the water separator, the main stream of air returns to the cabin, while a small portion is sent to the carbon dioxide system. Here, the remaining moisture is first removed in a silica-gel bed, then the carbon dioxide is removed by molecular sieves with the remainder of the air returning to the cabin. The carbon dioxide then goes to the reduction reactor where it is combined with hydrogen and is reduced to carbon and water, the latter going to the water storage tank.

The pure water storage tank receives water from urine recovery, the water separator, and the reduction reactor. Sufficient water is transferred to the oxygen generator, where it is electrolyzed into oxygen and hydrogen. The oxygen is injected into the cabin air, and the hydrogen is used in the carbon dioxide reduction reactor.

System Description and Analysis

The analysis of the system described requires calculation of a heat and material balance. The results of this calculation are shown in Figure 84. A revision of the space radiator duty (discussed in "Thermal Control") was necessary, because the electronic equipment heat load did not include the heat evolved in the molecular sieve and the oxygen generator. These factors are included in the following heat balance:

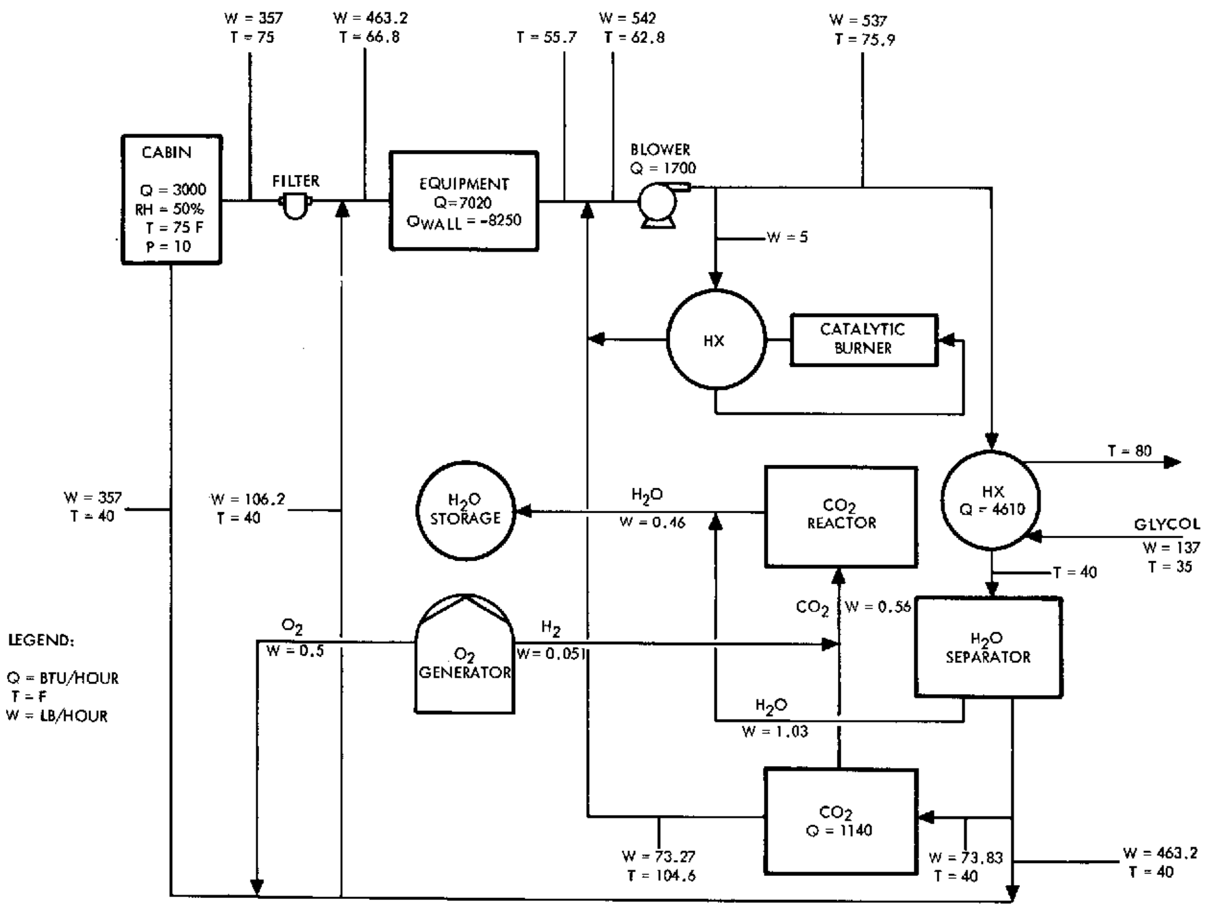


Figure 84. Heat and Material Balance Calculation

Item		Btu/hr
Equipment		6, 800
Molecular Sieve		1, 140
Oxygen Generator		1, 920
Metabolic Heat		3, 000
	Total	12, 860
Wall Heat Loss		8, 250
Space Radiator Load		4, 610

The heat balance shown in Figure 84 is for a particular set of conditions, as illustrated by the data in "Thermal Control." If the heat loss through the wall were zero, the radiator heat load would be 12, 860 Btu/hour. The heat balance for this condition has been calculated although not shown here.

Temperature Control

Temperature is controlled by cooling the air stream in a heat exchanger, using glycol as the cooling medium. The glycol is pumped through the heat exchanger, then into a space radiator, which rejects the heat picked up from the air. For the conditions shown on Figure 84, glycol flow is 137 pounds per hour, with a temperature range of 35 to 80 F. A bypass in the glycol loop around the heat exchanger provides variable control.

Humidity Control

Humidity is controlled by cooling the cabin air to its dew point. The condensed water is removed from the air stream in a centrifugal-type water separator. The air stream provides motive power for the separator, which, in turn, pumps the separated water to the water storage tank.

Oxygen Supply

In the system under discussion, the oxygen supply for breathing is supplied by an electrolysis cell. Electrolysis of water is a well-known process, and it is assumed that flight-type hardware will be developed by 1965. The system is desirable for extended duration missions; however, for the space laboratory, it should be pointed out that on the basis of six men per module, the power requirement is 1500 watts. At present, it would be difficult to provide sufficient solar arrays to achieve this amount of power in addition to the other requirements of the station. S&ID has, therefore, considered an alternate system using stored oxygen supplies on the basis that they will be replenished every 6 weeks.

Leak Makeup and Repressurization

As previously mentioned, the leak rate from the entire vehicle has been calculated to be about 18 pounds per day. For a 6-week period, this amounts to 756 pounds of gas composed of oxygen and nitrogen in the following proportions:

Gas	Proportion (percent)	Weight (lb)
Oxygen	34	256
Nitrogen	66	500
Total:	100	756

These gases will be stored separately as supercritical gas.

For repressurization of the vehicle, it was decided to provide enough stored gas to repressurize each module, the center hub, and the spokes twice during the mission. The gas requirements and tankage are

Item	One Module (lb)	Hub (lb)	Spokes (lb)
Oxygen Gas	140	77	103
Oxygen Tank	350	193	257
Nitrogen Gas	264	145	195
Nitrogen Tank	552*	300	410

*Two tanks

Pressure Control of Air in Radial Spokes

The air in the radial passageways between the hub and the ring modules normally will be stagnant, since the doors at each end would be closed to minimize leakage pressurization losses. Air will be circulated by the movement of crew members through the tube at times when the tube would be used for access passage between the central and radial modules. At other times, however, the air would be activated only by the artificially induced radial-gravitation force and the density variations due to temperature. The air temperatures would be determined by the surface temperatures of the tube structure.

As illustrated in Figure 85, a small valve to supply an air-jet flow through a nozzle could be provided to direct air from the central module conditioning system to produce a flow circulation in these spokes. Return air flow could be through a valve to the central module. An absolute pressure override would be included to close these valves in the event of depressurization of one of the spokes or of the central compartment.

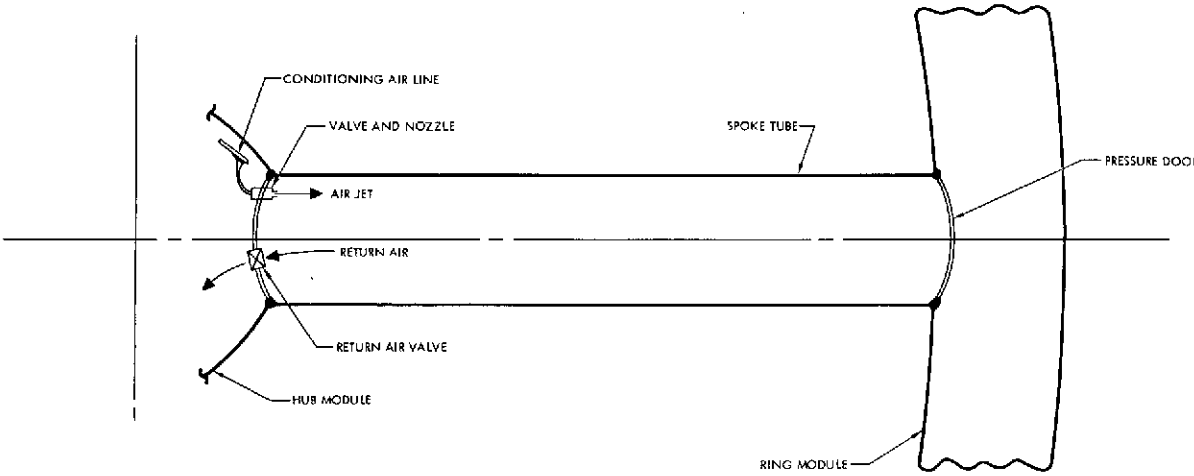
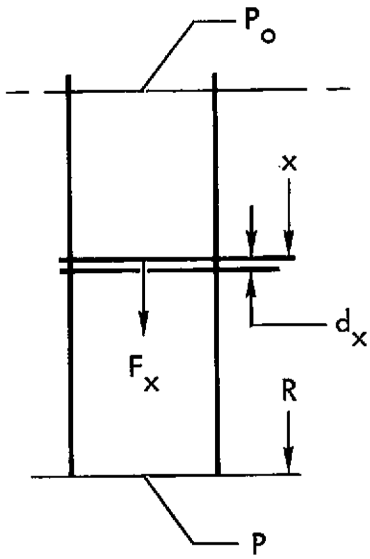


Figure 85. Radial Spokes Pressure Control System

The radial acceleration will produce an increase in pressure at the rim end of the tube. Temperature changes, also, will cause the more dense colder air to flow toward the rim end. These factors would indicate that the rim location would be better for the air jet nozzle at the expense of adding the conditioning requirement to each of three of the rim module systems.

Air Column Pressure Due to Radial Acceleration

The air column pressure due to radial acceleration is illustrated in the following diagram:



P_o - pressure at R = O

ρ_o - air density at R = O

X - radius distance from axis of rotation

Force due to radial acceleration:

$$F = ma$$

Mass of air column per square foot of area:

$$m = \rho X$$

Acceleration due to rotation:

$$a = \omega^2 k$$

Air total pressure at radius R

$$P_R = P_o + \int_o^R m\, a\, d_x$$

$$P_R = P_o + \int_o^R (\rho X)(\omega^2 X)\, d_x$$

$$P_R = P_o + \rho\omega^2 \int_o^R X^2\, d_x$$

$$P_R = P_o + \rho\omega^2 \left[\frac{1}{3} X^3\right]_o^R$$

$$P_R = P_o + \rho\omega^2 R^3 \left(\frac{1}{3}\right)$$

for:

$$P_o = 7.0 \text{ psi}$$

$$\rho_o = \frac{0.0765}{32.2}\left(\frac{7.0}{14.7}\right) = 0.00113 \text{ lb sec}^2/\text{ft}^4$$

$$R = 70 \text{ ft}$$

$$\omega^2 = \frac{32.2}{75} = 0.43 \text{ radians/sec}^2$$

$$P_R = 7.0 + \frac{1}{3}\left(\frac{1}{144}\right) (0.00113)(0.43)(70)^3$$

$$P_R = 7.385 \text{ psi}$$

Carbon Dioxide Removal

The selected carbon dioxide removal system consists of two silica-gel beds and two molecular sieve beds with appropriate piping and switching valves. The complete subsystem for six men would weigh 64 pounds. The silica gel is provided to remove water vapor, since the molecular sieves have a preferential affinity for water vapor over carbon dioxide. A portion of the main airstream enters the subsystem, passing first through one of the two silica-gel beds, which dries the air to a dew point of -100 F. The air then passes to a molecular sieve bed, in which carbon dioxide is absorbed. By proper valving, the air then enters the other silica-gel bed, desorbing, with the aid of electrical heaters, the water previously absorbed; the air then returns to the cabin.

The second molecular sieve bed, meanwhile, is connected to a vacuum pump which desorbs carbon dioxide and sends it to a storage tank, from which it goes to the carbon dioxide reduction reactor. For the alternate system using stored oxygen supplies, the molecular sieve is connected to the vacuum of space for desorbing.

The subsystem is controlled by a timer to provide the proper cycling of alternate adsorption and desorption of the beds.

Carbon Dioxide Reduction

In a regenerative atmospheric control system, oxygen must be recovered from carbon dioxide. At present, the most promising system for accomplishing this appears to be catalytic reduction of carbon dioxide with hydrogen according to the following equation:

$$CO_2 + 2H_2 \rightarrow C + 2H_2O$$

The water is then electrolyzed to oxygen and hydrogen, used for breathing and the catalytic reaction respectively. For a catalytic reaction of this type, the relationship between the weight of catalyst and the feed rate may be expressed by the following equation:

$$\frac{W}{E} = \int_o^X \frac{dX}{r}$$

where

W - weight of catalyst

F - feed rate

x - degree of conversion

r - reaction rate

The reaction rate, r, is difficult to evaluate; however, some data were found resulting in the equation

$$\frac{W}{F} = \frac{1}{kP\,5/2} \int_o^x \frac{\left[1 + PK\,(1-x)\right]^3 dX}{\sqrt{X(1-X)^2}}$$

where k and K are velocity constants, and P is the total pressure. This equation was solved with the IBM computer, using values of k and K at four temperatures, and P equal to one atmosphere. The results are illustrated in Figure 86. The equilibrium values of x at the indicated temperatures (Figure 86) were determined from previous calculations of the thermodynamic equilibrium for the chemical reaction. The corresponding values of W/F were found in this approach, and, knowing the required feed rate F, W was calculated. Using a catalyst bulk density of 94 pounds per cubic foot, reactor volumes, diameters, and lengths were calculated. Mass velocities and pressure drops were then determined. The results are shown in the following listing:

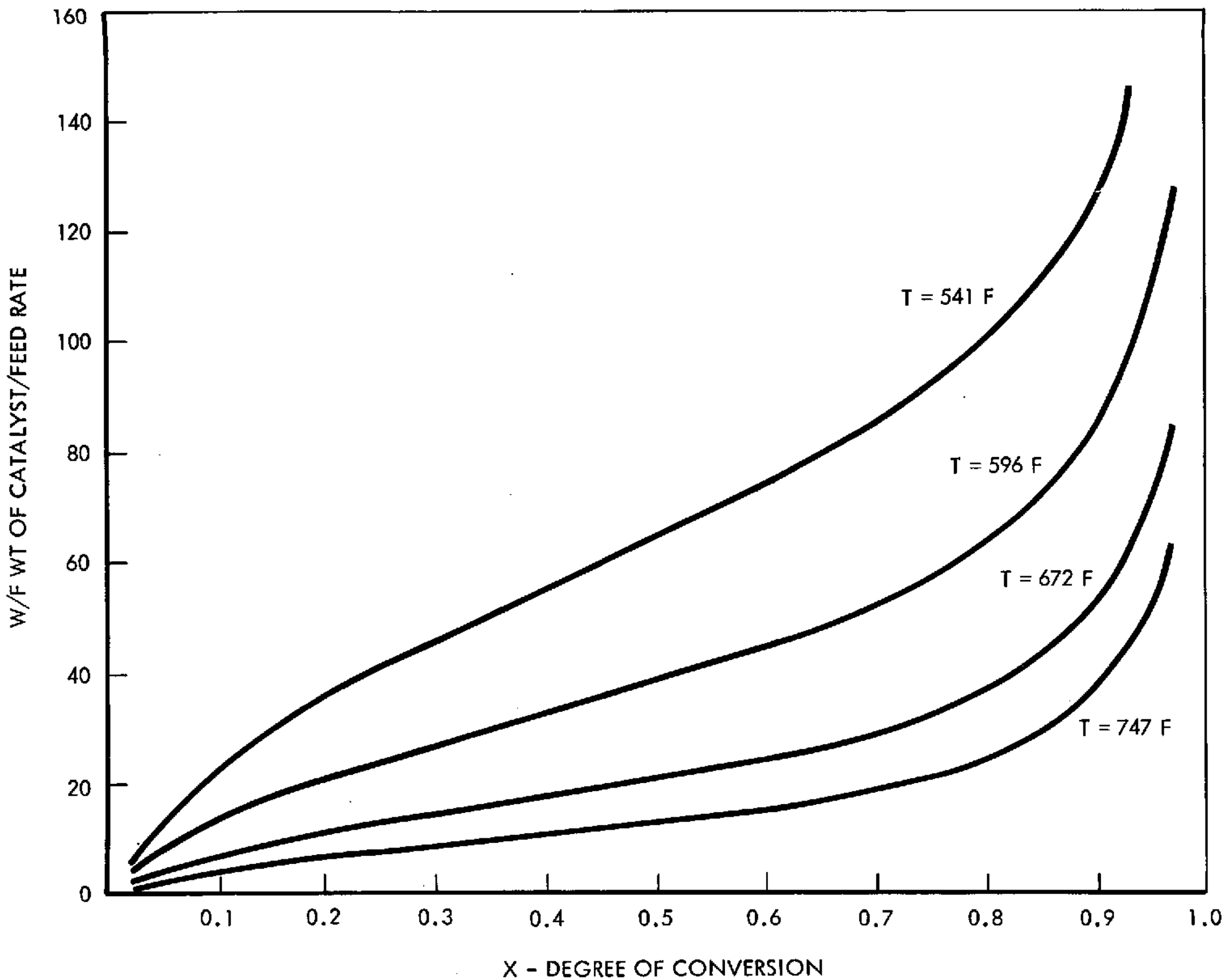


Figure 86. CO_2 Reduction

Temp (F)	Equilibrium x	$\frac{W}{F}$	Weight (lb.)	Volume (ft^3)	Diameter (ft)	Length (ft)	Mass Velocity	$\frac{\Delta P}{\Delta P\ 541}$
541	0.97	180	120	1.28	0.74	2.96	2.05	1
596	0.94	103	68	0.73	0.62	2.46	3.02	1.91
672	0.89	50	33	0.35	0.48	1.92	5.08	4.52
747	0.82	26	17	0.18	0.38	1.52	8.33	10.3

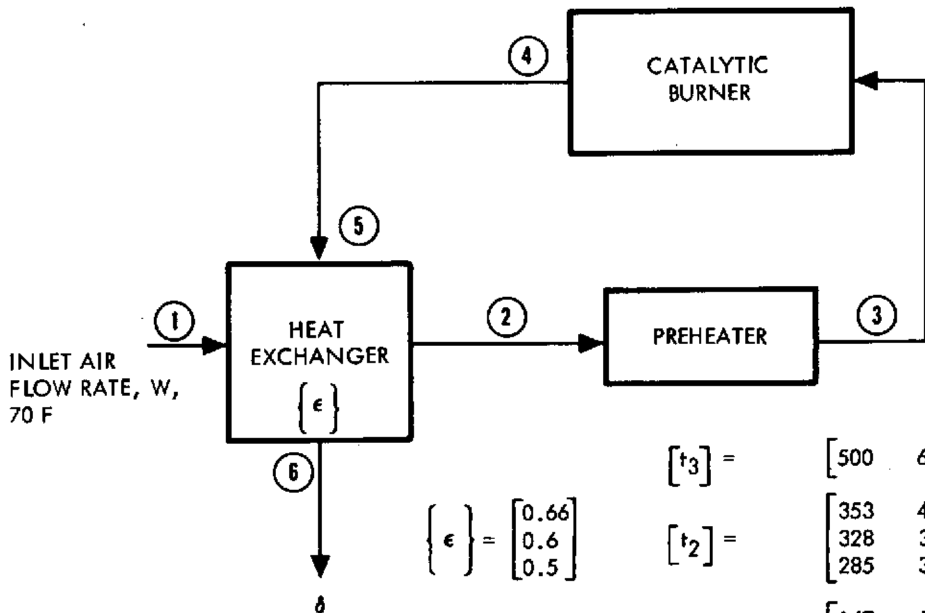
The last column in the listing shows the pressure drop relative to that at 541 F, a factor directly affecting blower power requirements. Plotting W and ΔP, versus temperature, yields two curves crossing at an optimum point of 649 F. This would give a reaction conversion of 92 percent of the carbon dioxide, using about 35 pounds of catalyst in a reactor 6 inches in diameter by 2 feet long. Since the reaction is exothermic, electrical heating requirements should be negligible once the reaction has started.

Contaminant Removal System

Contaminants of a particular nature (e.g., dust) can be removed by circulating air through a filter. To minimize dust dispersion throughout the module, the air-filtering intake should be close to the source of the dust or other contamination. Odors and many vapors are adsorbed by activated charcoal, which is capable of adsorbing up to 50 percent of its weight in higher boiling organic materials (Reference 1).

A system such as that shown in Figure 87 will be required to reduce concentrations of combustible gases and vapors. One material successfully used in a catalytic combustion process is Hopcalite—basically a mixture of manganese and copper oxides (Reference 2). The efficiency of contaminant removal with this catalyst is determined by the nature of the contaminant and the operating temperature of the catalyst bed. Hydrogen and most hydrocarbons are removed very effectively (over 90 percent) at temperatures in the 500 to 600 F range. Methane reacts much more poorly, requiring a temperature of 700 F to obtain a removal efficiency of 30 percent (Reference 1).

The size of the catalyst unit will be determined by the air flow rate. The recommended minimum for Hopcalite is one cubic foot per minute/square inch of area (Reference 2).



$$\{\epsilon\} = \begin{bmatrix} 0.66 \\ 0.6 \\ 0.5 \end{bmatrix}$$

$$[t_3] = \begin{bmatrix} 500 & 600 & 700 \end{bmatrix} \text{ F}$$

$$[t_2] = \begin{bmatrix} 353 & 420 & 485 \\ 328 & 388 & 448 \\ 285 & 335 & 385 \end{bmatrix} \text{ F}$$

$$[\Delta t_{2-3}] = \begin{bmatrix} 147 & 180 & 215 \\ 172 & 212 & 252 \\ 215 & 265 & 315 \end{bmatrix} \text{ F}$$

$$[t_6] = \begin{bmatrix} 217 & 250 & 285 \\ 242 & 282 & 322 \\ 285 & 335 & 385 \end{bmatrix} \text{ F}$$

$$[P_{2-3}] = \begin{bmatrix} 14.7 & 18.0 & 21.5 \\ 17.2 & 21.2 & 25.2 \\ 21.5 & 26.5 & 31.5 \end{bmatrix} \text{ WATTS}$$

(PER EQUIP KW)

BURNER EFFICIENCY > 90% AT
700 F IN HOPCALITE FOR MOST
HYDROCARBONS EXCEPT METHANE

Figure 87. Catalytic Combustion System for Contaminant Removal

The system for the catalytic combustion of the combustible contaminants, as shown in Figure 87, normally will extract a portion of the air from the thermal and atmospheric control system to minimize the complexity of the system installation. In this manner, a fraction of the air circulated through the conditioning system is the air supplied to the contaminant control system. Providing a flow of 1 percent of the conditioning supply would have the following effects on the system:

The cooling air required for equipment cooling per kilowatt of power dissipation with a temperature rise of 100 F is

$$Wc = \frac{3413}{0.24\ (100)} = 142 \text{ lb/hr kw}$$

One percent of this flow rate would be 1.42 lb/hr. Values of temperatures throughout the system are listed in the Figure 87 for three values of heat exchanger effectiveness and three values of operating temperature for the catalyst bed. The heater power required to heat the 1 percent flow for a one kilowatt cooling load is also tabulated.

Comparing the suggested 1 percent flow rate from a 1 kilowatt cooling air circuit, with the requirement previously calculated for removal of methane, indicates that the methane requirement is 23 times greater (for a safety factor of 100) than the 1 percent of a 1 kilowatt cooling load. Higher values of conditioning load would reduce this difference.

The reaction products included in the general equation for the system may be indicated by the reaction of methane with oxygen occurring in the catalytic burner. The chemical reaction for the methane burned is

$$CH_4 + 2O_2 \rightarrow CO_2 + 2H_2O$$

One volume of methane combines with two volumes of oxygen from the air to produce one volume of carbon dioxide gas and two volumes of water vapor. The weight proportions of the reactants is 4 parts oxygen to 1 part methane, 2-3/4 parts carbon dioxide, and 2-1/4 parts water. For most of the contaminant substances, these values will be very low percentages of the conditioning system quantities, but they may have to be considered when summing up all the contamination materials and the accumulation over extended periods of time.

WATER RECLAMATION

Assuming the average man requires approximately 2200 grams of water each day for ingestion, it is observed that long duration missions will require an additional 1000 grams for daily requirements in food, reconstituted juices, etc. The recovery of potable water from human waste becomes very desirable for missions longer than several days.

The water excreted from the lungs and skin of a man and subsequently condensed by a vehicle's environmental control system has undergone a change of phase and can be made potable by adsorption filtration and ion exchange processes. The recovery of potable water from urine and wash water is more difficult because the concentration of contaminants is approximately 4 percent. The recovery of potable water from untreated fecal matter is very difficult but practical when combined with other wastes subject to water recovery. Assuming dehumidification use, and that such dehumidification of the cabin gas produces 1000 grams of water per man-day, a water recovery system is expected to produce 2200 grams of potable water each man-day from an estimated 2400 grams of urine and wash water. The efficiency of a water recovery system should, therefore, be greater than 92 percent.

American Machine and Foundry Company has developed a prototype water recovery system for LRC that was designed to recover 650 grams of water an hour in a weightless state (AMF Model MR 08-062). When operated for 24 hours each day the system shows a capability of supporting seven men. The requirements satisfied in the construction of this unit are as follows: (1) quality of recovered water conforming to the Public Health Service and to drinking water standards of 1946, (2) recovery efficiency of at least 92 percent, (3) leakage of ambient gas not exceeding 0.05 pounds per day, (4) power shall be less than 150 watts, (5) weight shall be less than 45 pounds, (6) volume shall be less than 1.9 cubic feet, and (7) the process by compression distillation.

The salient features of this prototype unit are as follows:

1. The electric motor and Roots-type compressor are located inside the unit to minimize system volume and also to utilize waste heat for elevating the operating pressure levels.

2. The evaporator bowl is lined with a removable plastic bag to facilitate the removal of urine residue. This bag is sucked against the evaporator wall by evacuation.

3. The evaporator bowl rotates so that, under weightless conditions, centrifugal action forces the waste liquid against the evaporator wall and aids in the removal of water from the condenser surface.

4. The compressor and evaporator bowl are both driven by one electric motor through a transmission.

5. Condensate can be forced out of the unit and through an adsorption filtration cartridge by simply repressurizing the system with the compressor running. Overload protection of the motor is afforded by a relief valve between the condenser and evaporator.

6. A view port and transparent evaporator cover are provided so that the waste liquid can be observed during system operation.
7. An internal valve is provided to close the compressor suction line and avoid entrainment of waste liquid during fill operations or abrupt changes in vehicle attitude.

A block diagram of a typical waste water recovery system is shown in Figure 88. In addition to the compression distillation unit, a water purification unit must be added to the system, since the distillation unit does not remove bacteria from the water. The relationship between the various units is illustrated in the figure.

INTEGRATED SYSTEM INSTRUMENTATION

The piping and instrumentation diagram for the thermal and atmospheric control system is shown in Figure 89. Essential instrumentation of the system is described in the following discussion.

The main blower is equipped with a pressure switch that measures the difference in pressure across the suction and discharge. The pressure switch will activate an alarm if the pressure drop decreases below a preset value. The spare blowers can then be activated while the trouble is being investigated. Blower discharge pressure and flow rate are measured at the discharge.

The air flow rate of the air-glycol heat exchanger is set and controlled by a flow controller operating a valve which has a manual hand jack for manual control. Flow rate indication is provided on the control panel.

Flow rate through the glycol loop is controlled in the following manner. The temperature of the outlet air from the air-glycol heat exchanger triggers the temperature controller, which in turn positions the control valve in the radiator, thus permitting additional cooling. The glycol pumps are equipped with a relief valve and pressure gage. The flow rate is indicated by a sight glass in the line.

Air rate to the carbon dioxide removal system is set by a hand control indicating valve; the flow rate is indicated on the control panel. Carbon dioxide and hydrogen are sent to the carbon dioxide reduction reactor under control of a ratio flow controller, which properly proportions the two gases. Temperatures in the reactor are measured by thermocouples with read out on a multipoint recorder at the control panel. (See Figures 90 and 91.)

Oxygen supply for breathing is added under pressure control upon demand as measured by the oxygen partial-pressure sensor.

A total-pressure sensor provides the input to a pressure controller on the nitrogen tanks, thus admitting nitrogen gas flow to maintain total pressure and leakage makeup. Since leakage also removes oxygen, as well as nitrogen, a small amount of makeup oxygen is continuously added from the stored oxygen tank under flow control. The rate can be set proportionally to the required nitrogen flow for total pressure control, since actual leak rates are unknown.

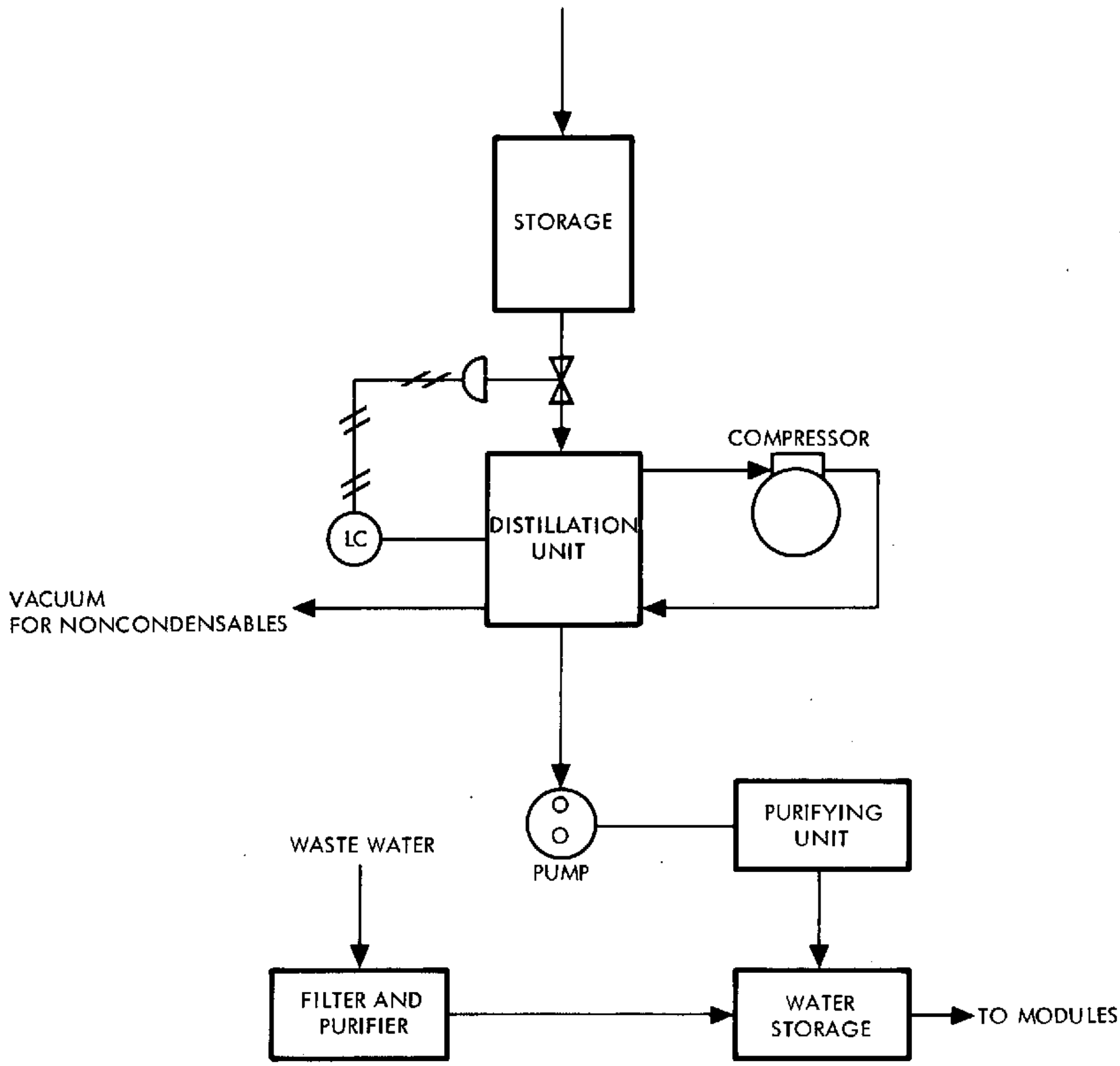


Figure 88. Waste Water Recovery System

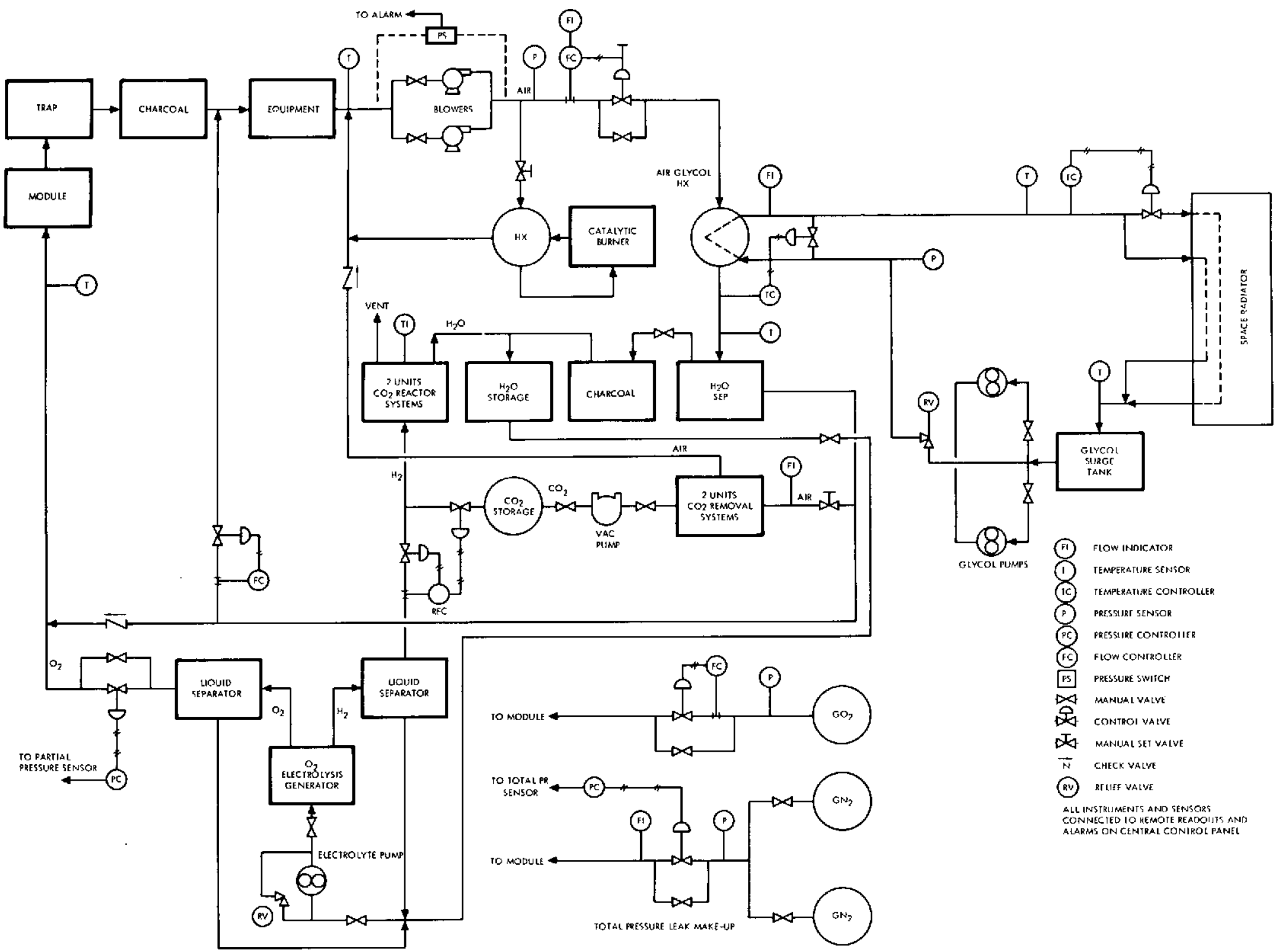
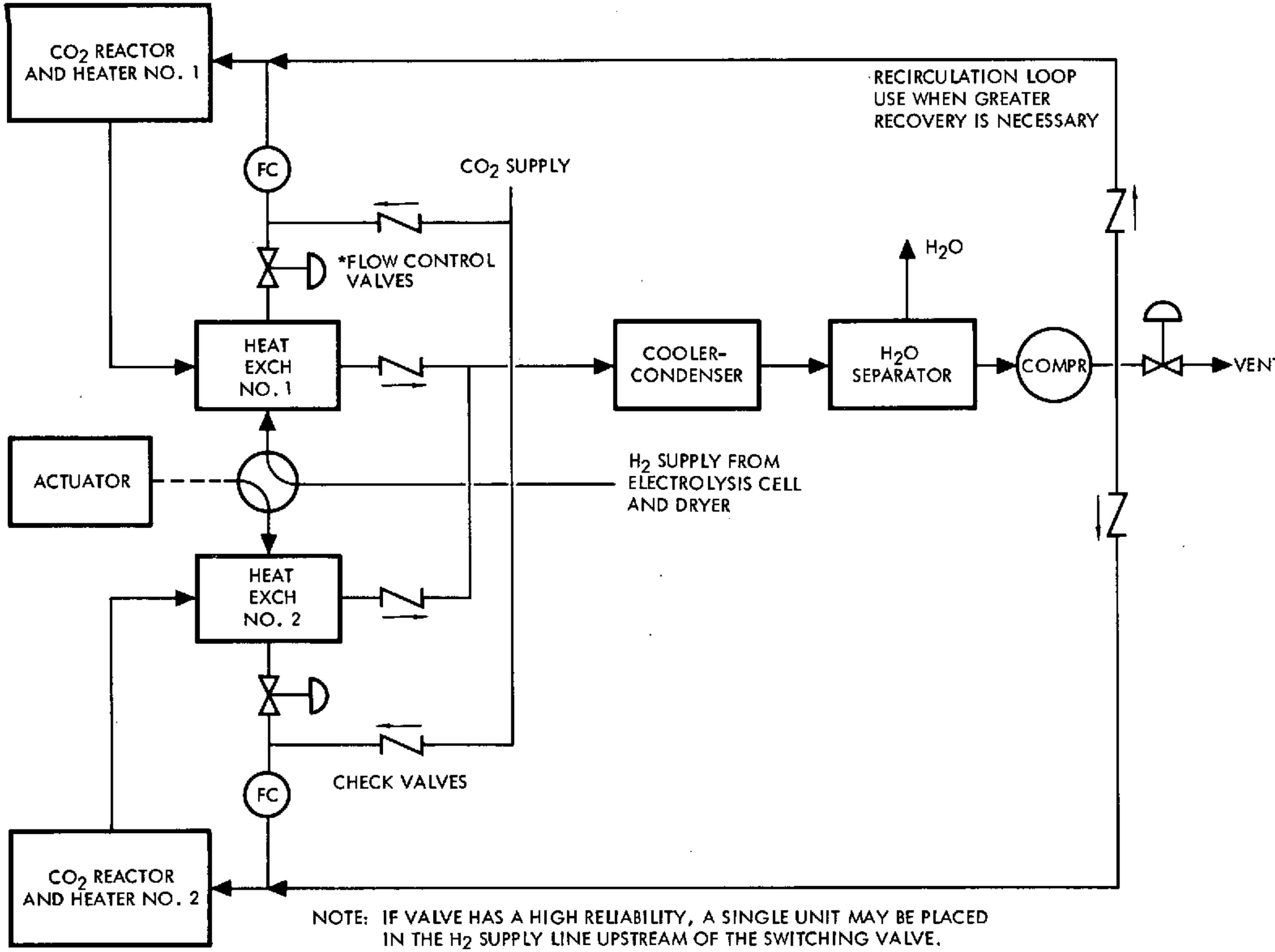


Figure 89. Thermal and Atmospheric Control System - Piping and Instrumentation Diagram



NOTE: IF VALVE HAS A HIGH RELIABILITY, A SINGLE UNIT MAY BE PLACED IN THE H_2 SUPPLY LINE UPSTREAM OF THE SWITCHING VALVE.

Figure 90. Catalytic CO_2 Reduction System - Hydrogenation System

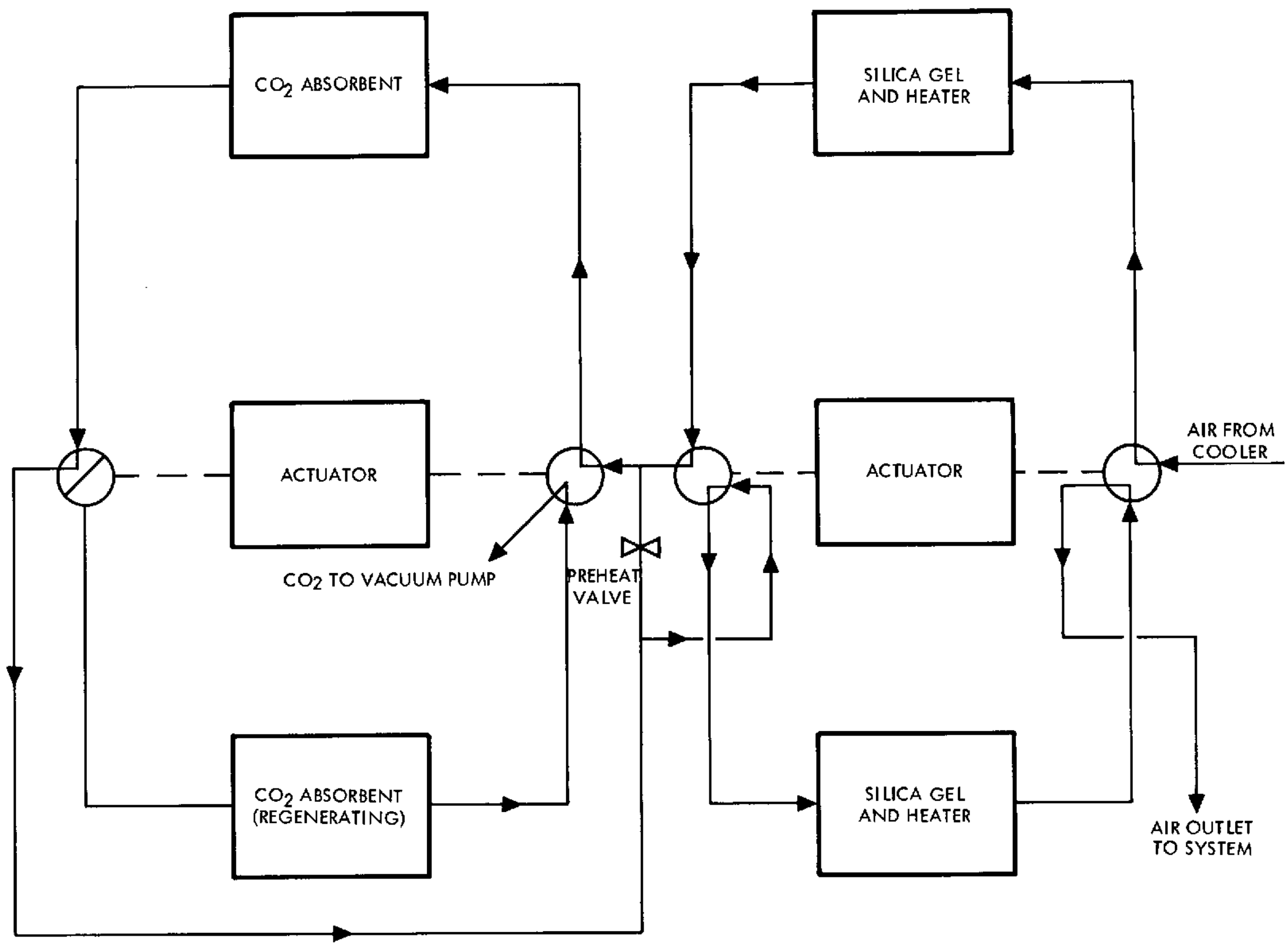


Figure 91. Regenerable CO_2 Removal System Schematic

The control panel for the system should contain the following gages and indicators:

1. Gas analyzer of the mass spectrograph type which continuously samples the atmosphere for oxygen, carbon dioxide, hydrogen, hydrogen sulfide, carbon monoxide, and methane.

2. Multipoint temperature indicator, into which all temperature measurement points are connected.

3. Gages reading humidity, partial pressure of oxygen and of carbon dioxide, total pressure, and module temperature. These gages are provided with sensors separate from those in system piping.

4. Alarm indicators for low oxygen, high carbon dioxide, low total pressure, low air flow, high glycol temperature, and other critical items.

WEIGHT AND POWER SUMMARY

A comparison of the system weights and power requirements for a stored oxygen system and a regenerative oxygen system are given in Table 11. The figures are for one module. As mentioned previously, S&ID had considered the stored system with six weeks resupply and, on the basis of early discussions with NASA, it was agreed to use stored oxygen, and employ the space station as a test bed for the less-developed regenerative systems. More recent discussions indicate a preference for the regenerative system, and therefore the comparison is presented in Table 11.

WASTE MANAGEMENT

In a separate research program, S&ID has designed and constructed prototype components of a waste arrangement system applicable to multi-manned space vehicles with relatively long mission durations. The concept—utilizing separate waste collection and waste storage units—provides a means of handling many waste products generated aboard the space station. Since the concept was described in detail in the midterm report, it is not repeated herein.

PERSONAL HYGIENE

In conjunction with the S&ID study, the Whirlpool Corporation has conducted a separate analysis of personal hygiene requirements for the space station. While the following concepts presented are those of S&ID, it is emphasized that the Whirlpool Corporation concepts (discussed in detail in their supplemental report) may be equally applicable; final determination

Table 11. Comparison of Weights and Power Requirements for Stored Oxygen and Regenerative Oxygen Systems

Item	System	
	Stored Oxygen	Regenerative Oxygen
ECS System Equipment and Gas Tankage	1415	1417*
Instruments and Valves	41	41
Water Recovery Subsystem	59	59
Expendables:		
1. Oxygen, Breathing	504	
2. Oxygen, Leak	140	140
3. Nitrogen, Leak	264	264
Total Weight (lbs.)	2423	1921
Power (watts)	801	2351

*Does not include weight of additional solar array, batteries, and charger for electrolysis of water to produce oxygen.

of optimum approaches and techniques is dependent upon further analysis and evaluation. No question exists, however, as to the ability to establish practical concepts compatible with overall requirements and operational time period.

Personal hygiene considerations include all concepts normally considered for life on earth. As near as is practicable, all personal hygiene activities will duplicate those to which the crew are accustomed.

Facilities are provided for such normal personal hygiene practices as body bathing, hand and face cleansing, oral hygiene, shaving, hair-cutting, finger- and toe-nail care, and clothing hygiene. A dispensary for limited emergency medical care is also provided.

Three areas are outlined and discussed: (1) a shower with an adjacent dressing area; (2) a lavatory for hand and face cleansing, oral hygiene, shaving, etc.; and (3) a dispensary. (Water conservation and handling will be defined in a separate discussion.)

Shower and Dressing

The shower and dressing area (Figure 92) is arranged adjacent to the lavatory for convenient access to all personal hygiene facilities. A recirculating shower system, with an approximate capacity of 8 gallons is provided for body cleansing, using a controlled amount of water. A shower cycle is programmed as follows: The crewman turns on the master switch (a) in the control panel (Figure 93). Indicator light (b) will glow if the shower system is ready for operation. Next, the crewman selects the water temperature on temperature selector (d) and presses switch (e), which closes the circuit to immersion heaters (h). Heating of the water will require approximately 20 minutes. When the selected water temperature has been reached, indicator light (f) will glow. The crewman then steps into the shower (Figure 94) and closes the door. The single selector lever (j) on the wall of the shower is placed to SOAK, causing a controlled quantity of water to flow when button (k) is pressed. Water will flow at approximately 1.8 gpm for 2 minutes, shutting off automatically. After soaping, the crewman places selector lever (j) to rinse and initiates a water flow by pressing button (k). The water used to soak with is now repumped from sump (r) through the shower spray (p). The rinse cycle - using the same water - may be repeated by pressing button (k). A final clean rinse may be had by returning selector lever to SOAK and again pressing button (k). When showering and rinsing are completed, selector lever (j) is turned to the transfer position. All waste shower water will be transferred through coagulant and contact filters to water reclamation automatically when the master switch (a) is placed in the OFF position. Indicator light (c) will glow until the transfer is completed. When the transfer is completed, tank (g) will fill automatically. Drying of the body will be accomplished with lintless bath towels normally stored in a cabinet (Figure 95). (Towel-drying is necessary to aid in removing some of the waste skin tissues or keratin fibers.) When body drying is completed, the towel and wash cloth are placed in the towel locker, where gas flow from the environment control system will dry them by convection evaporation.

Lavatory and Laundry

The lavatory and laundry area houses facilities for personal toilet activities, such as hand and face cleansing, shaving, oral hygiene, etc. Cabinet storage space, linens, and garment laundry facilities are also in this area. Since water usage is a major function in this area, the physical relationship of the shower and lavatory is arranged to complement system needs relative to water conservation.

Two wash basins are arranged in a counter-high installation so that two crewmen may use the facilities at the same time. Water for personal hygiene is stored in a container of the same design concept as other on-board

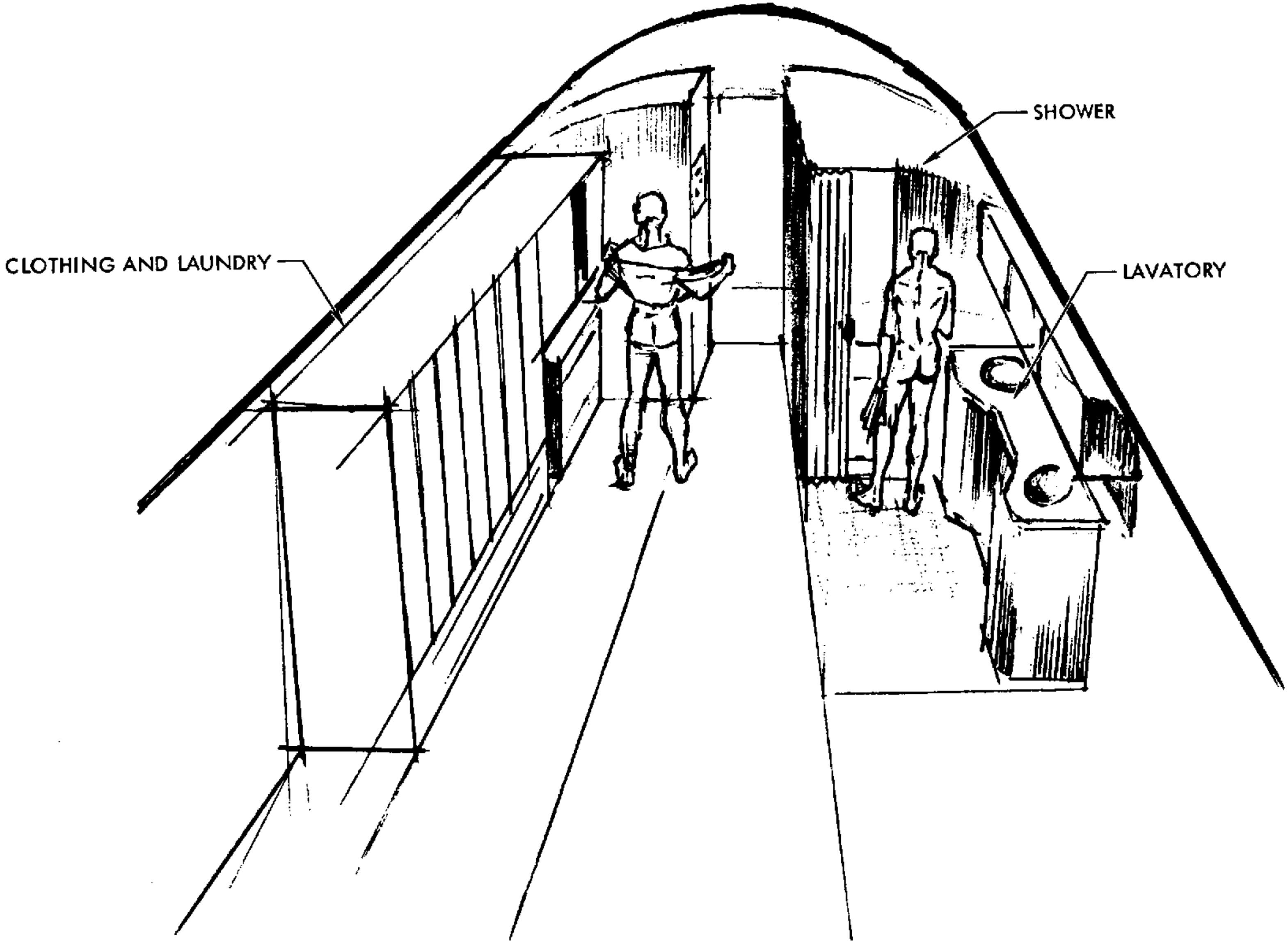


Figure 92. Shower and Dressing Area

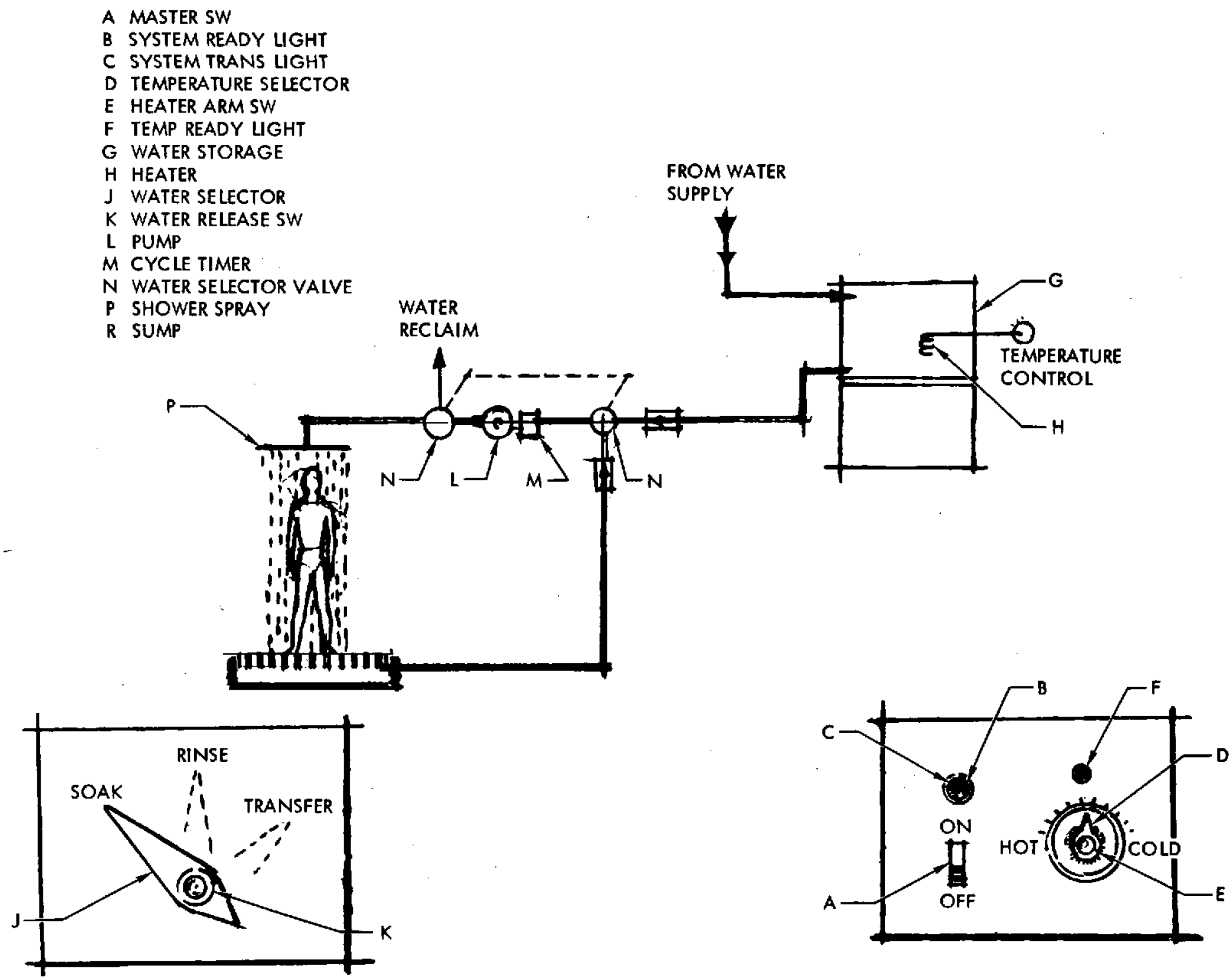


Figure 93. Shower Cycle

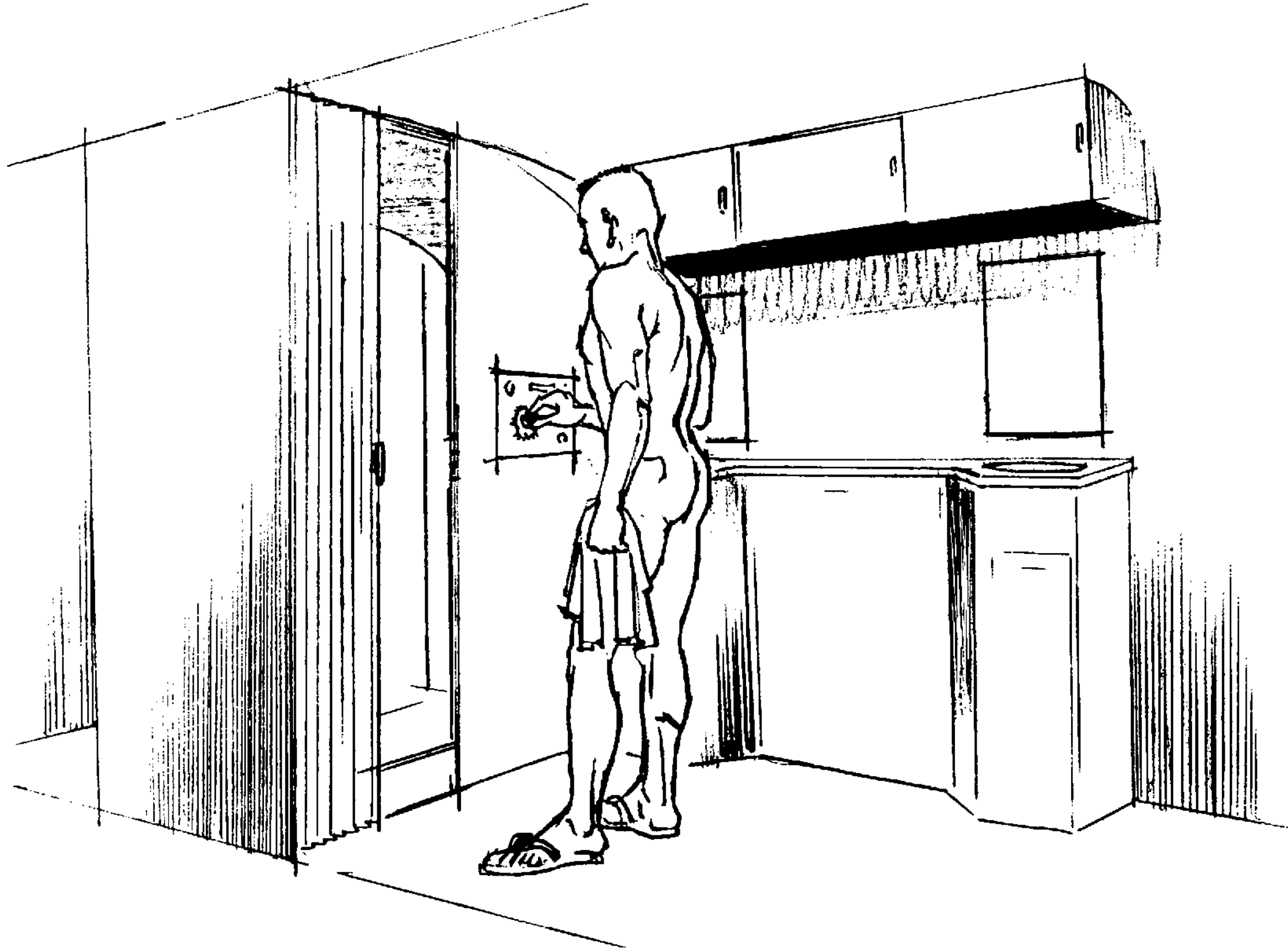


Figure 94. Shower Procedure

A CLOTHES LOCKER
B LAUNDRY
C TOWEL LOCKER
D SUPPLIES

Figure 95. Body Drying Procedure

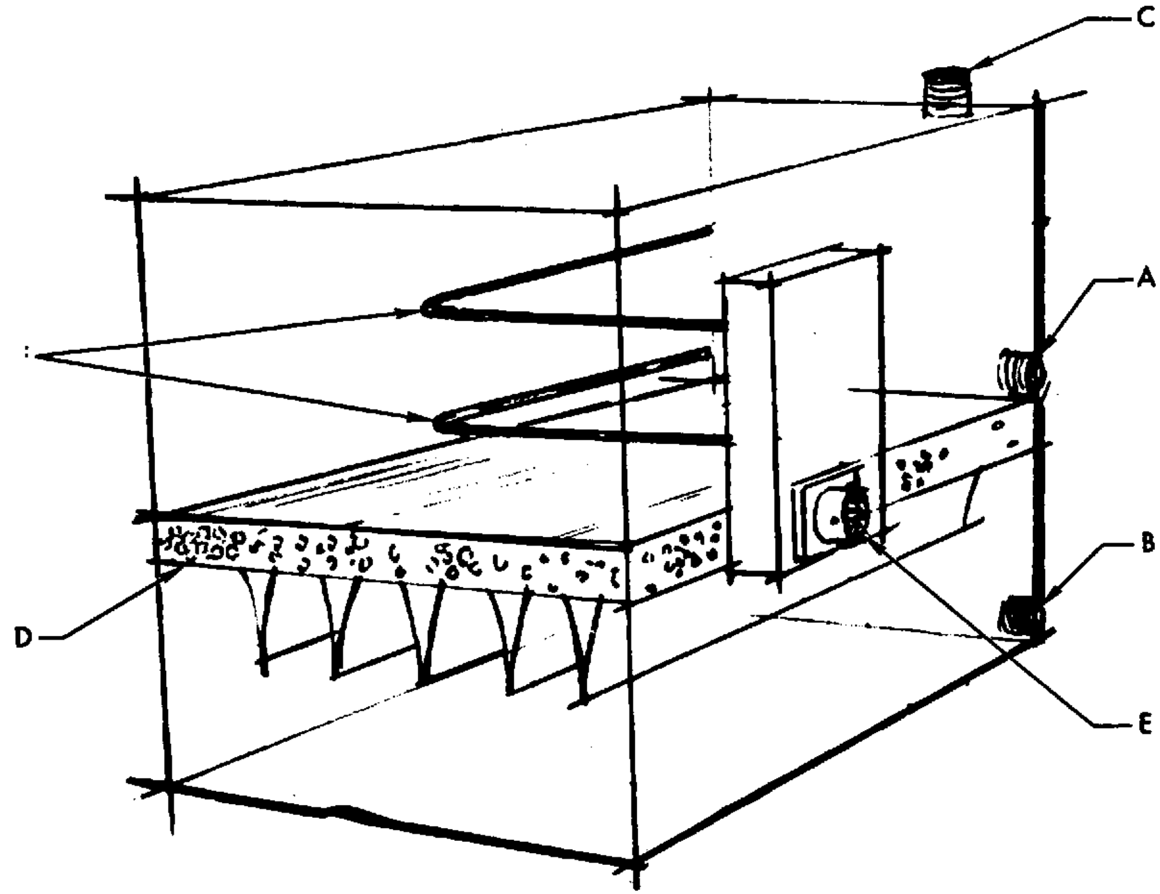
water storage containers (Figure 96), i.e., split tank with hot water occupying the upper portion. The total water capacity will be 2.5 gallons. Two gallons will be heated by two probe heaters — each at a different level— supplying rapid initial heating (approximately 12 minutes). Temperature is maintained by a thermal electric conversion unit located in the liquid-dividing wall. The control panel, similar to the control panel for the food preparation system, will permit one or two crewmen to select water heating levels (refer to "Food Preparation System"). For example, if one crewman is to use the facility, only 1 gallon of water will be programmed for heating. If two men are using the facility, 2 gallons will be programmed. The cold water level in the lower section of the tank fills automatically. Cold water (approximately 0.5 gallons) is made available to the basin separately from the heated water and is used for oral hygiene, etc. In front of and above each basin are a mirror and a cabinet for storing basic toilet items and equipment. The cabinet is vented to the environmental control system.

A special lanolin or compatible oil liquid detergent will be used for all skin-cleansing. This will be dispensed from a device housing a soap cartridge. This cartridge will hold a sufficient volume of liquid soap to last for the duration of a resupply cycle. (Liquid soap is recommended because it does not pose the same waste problem as that of bar soap.)

The cabinet space above the basins will contain several specific storage areas and compartments, all vented to the environmental control system. One compartment will contain each crewman's toothbrush and oral detergent. Another cabinet compartment will contain lotions, powders, and creams. Space for a shaving instrument will be provided in each cabinet at each basin. This shaving instrument will be operated mechanically or electrically; its design facilitates smooth shaves with minimum skin abrasion and irritation. (The clipped whiskers will be retained in the instrument in a special compartment.)

Hair-cutting will be done in the lavatory area. The hair-cutting equipment will be stored in the cabinets above the basins. The clippers are designed to permit the collection of cuttings in a container within the hair clipper. A storage compartment in the cabinet below the basins is fitted with a portable, disposable waste container. This container, when fitted, is vented to the environmental control system. After the container has been filled, it will be stored inside the daily-food waste container at the end of each day's duty cycle.

A clothing storage and laundry are provided adjacent to the shower and personal hygiene area. The clothing storage consists of several lockers and storage drawers. Next to these lockers is a special laundry unit and personal towel locker. Storage cabinets above the laundry unit are provided for



A HOT WATER OUT
B COLD WATER OUT
C SUPPLY WATER IN
D THERMAL ELECTRIC CONVERSION UNIT
E POWER CONNECTION
F HEATER ELEMENTS

Figure 96. Personal Hygiene Water Storage

utility cleaning materials such as degreasing agents, janitorial supplies, etc. The laundry unit design and operation most likely will incorporate some type of mechanical and electrical closed system. It is likely that all clothing will be manufactured from inert materials, making sterilization unnecessary. Where special clothing or soft goods require sterilization, exposure to the space vacuum will probably provide sufficient sterilization.

Dispensary:

The controlled environment within the space station will tend to reduce many simple communicable diseases as well as to aid in checking undesired bacteria carried in the air. General medical care and simple surgical techniques must be considered. To what extent such care will be planned for requires a more detailed study beyond the scope of this report. This section is limited to area allotment with reference to minimum medical and surgical care.

A dispensary will be provided for semi-isolation of injured personnel adjacent to the sick bay, as illustrated in Figure 97. The dispensary is arranged (Figure 98) to allow treatment of ambulatory or confined patients. The treatment of a patient may require him to be lying down on an examination or treatment table (a). A folding seat with an examination light (b) is provided for convenience in examination of ambulatory patients. The work area (c) of the dispensary contains a sink or wash basin with a temperature-controlled water system similar to the lavatory water system. The lower cabinets in the sink area are compartmented for bulk storage of medical supplies. The lower cabinet contains a surgical instrument sterilizer at deck level, and storage area for surgical instruments. The upper cabinets (d) contain a locked drug cabinet, solution and medicant storage, bandages, etc. Supplies for limited dental care will also be housed in these cabinets.

Equipment Weights

The approximate weights for the personal hygiene equipment for a 6-week mission for 21 men are listed below.

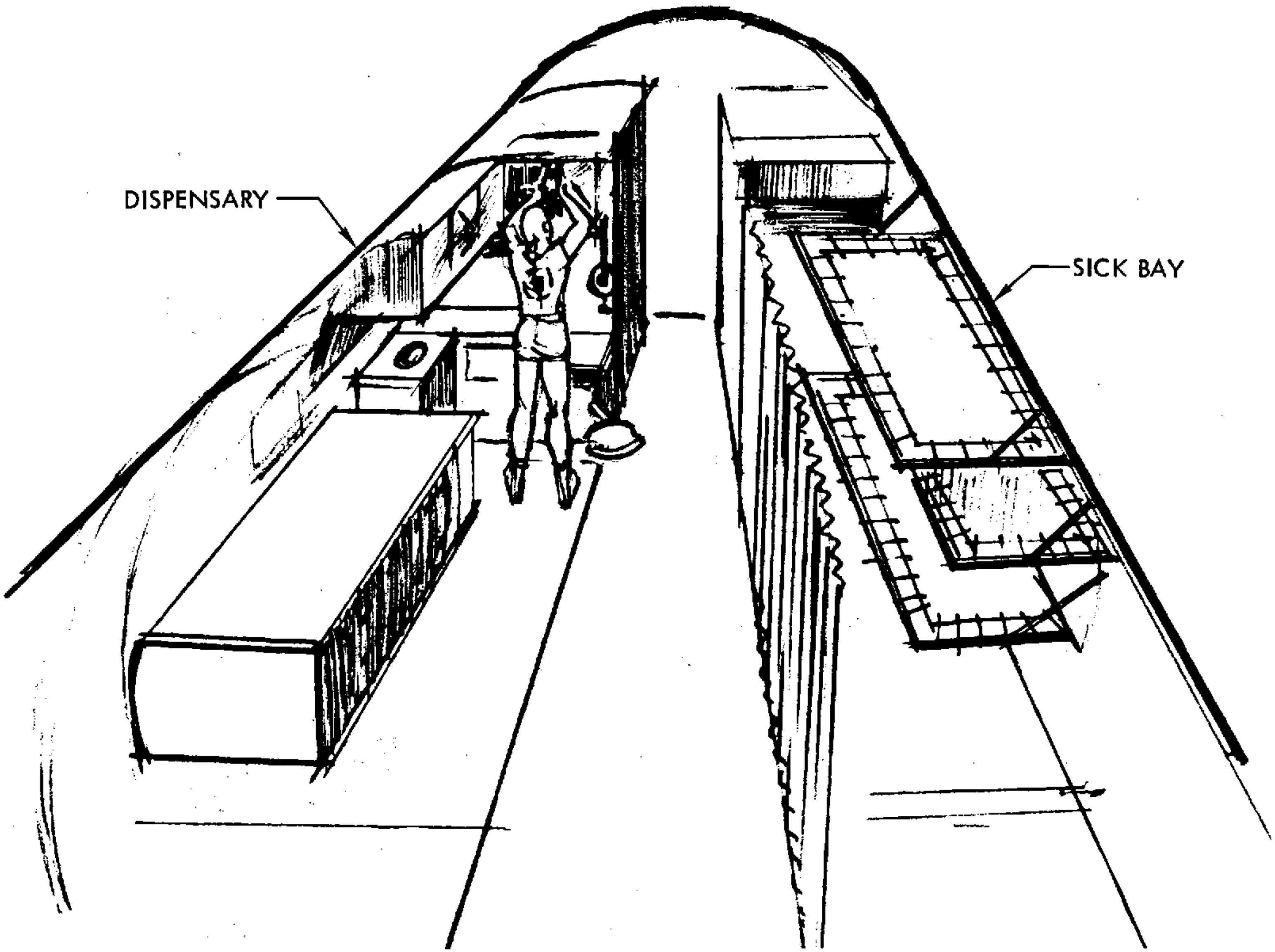


Figure 97. Sick-Bay Arrangement

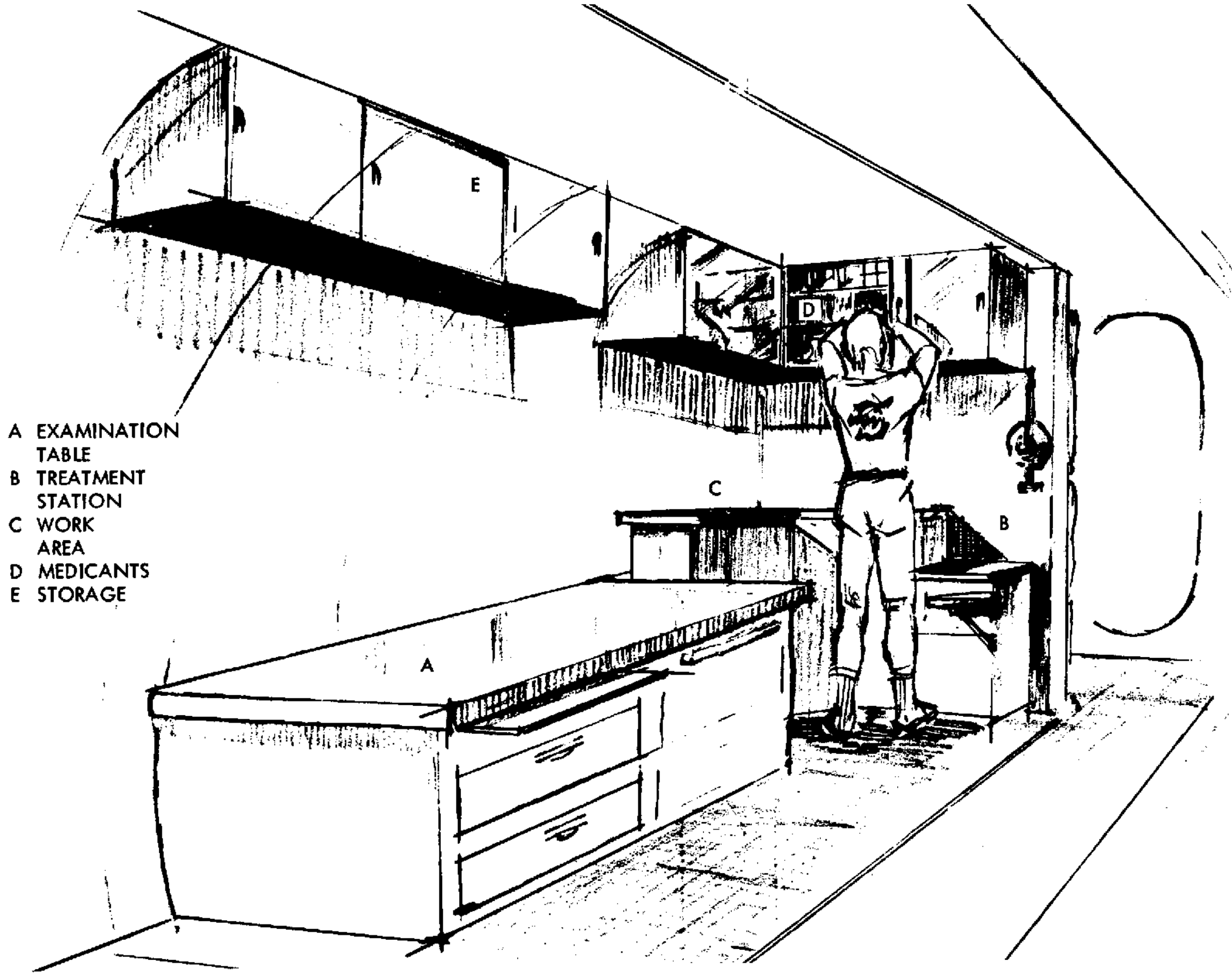


Figure 98. Dispensary Arrangement

Equipment	Weight (lb)
Shower	35.5
Lavatory	49.5
Dispensary	21.0
Clothing (garment change every two weeks)	37.8
Shoes, two pair	25.2
Water Required:	
1. Shower	192.0
2. Lavatory	60.0
3. Dispensary	32.0

FOOD STORAGE AND PREPARATION

While not a factor directly contributing to the space station feasibility examination, food storage and preparation is one of the important elements which must be considered in the design of a large multimanned vehicle. The food preparation and storage concepts discussed in this section were incorporated in the module internal arrangements previously described in "Configuration Analysis." In conjunction with the S&ID studies, Whirlpool Corporation has conducted a separate analysis of the food preparation and storage equipment. Although the following concepts are those of S&ID, the Whirlpool Corporation concepts (discussed in their supplemental report and submitted separately) may be equally applicable.

Food Preparation

Food rations will be packaged and wrapped as individual meals. Complete meals will then be group-packed in a combination food/waste storage container. This container will supply one duty crew (seven crewmen) with food rations for one day. When the container is unsealed and the food transferred to the food dispenser, the empty container is inserted in the lower part of the food preparation bar to become a waste receptacle. The food/waste container is a lightweight, rectangular-shaped, 8 by 8 by 24-inch pressure vessel, able to withstand its own support load during boost. The number of these vessels used is, of course, determined by the resupply cycle employed.

Each crew member serves and prepares his own meals at the food preparation bar shown in Figure 99. The crewman selects his food tray from the tray storage (a) and his eating utensils from his assigned locker drawer (b). He places his tray on the bar at (c) and selects food servings from a food pantry dispenser (d), which he installs on his tray. He then proceeds to prepare food and drink for consumption utilizing the hot or cold water dispenser spigots (f). With food tray in hand, the crewman moves to the adjacent seating area (Figure 100) where the tray is used as a table by attachment to a mating fixture. After eating, the crewman transports his tray back to the preparation bar (c), where he removes the individual food serving wrappings from the tray, inserting this waste in the food/waste storage container (e). All remaining garbage is cleaned from the tray, using chemically treated cleaning tissues (h) supplemented by water from spigots when necessary. Eating utensils are placed in an automatic washer (j) and cleaned. The cleaned items are then returned to their respective areas.

After the day's last meal, the waste-containing food/waste storage container is resealed and returned to its storage position, and the next day's food supply container is removed for installation in the food preparation area.

The food preparation water storage container reflects the same design concept as the other on-board water storage containers, i.e., a split tank with hot water occupying the upper portion as shown in Figure 96. The total water capacity will be 5 gallons, 3 of which will be heated by two probe heaters, each at a different level, supplying rapid initial heating (approximately 10 minutes). Sustained temperature is maintained by a thermal electric conversion unit located in the liquid dividing wall. The control panel will permit either four or seven crewmen to select water heating. That is to say, if four or fewer crewmen wish to eat at any one time, the water storage container will restrict the water level, allowing emersion of the lower probe heater element only. The cool or unheated water will refill automatically when the water level falls below the half-full mark.

The tray or food server illustrated in Figure 101 will provide three individual food mixing and retaining spaces: one for a removable drink mixing cup, one curved recess for eating utensils, and one rectangular recess for additional food stuffs.

The individual food wrappings are placed in the food mixing and retaining spaces and secured in place by hoop clamps (similar to embroidery hoops). The top or middle of the food wrapping is torn off and the food exposed for reconstitution. Drink mixes are similarly handled by the drink

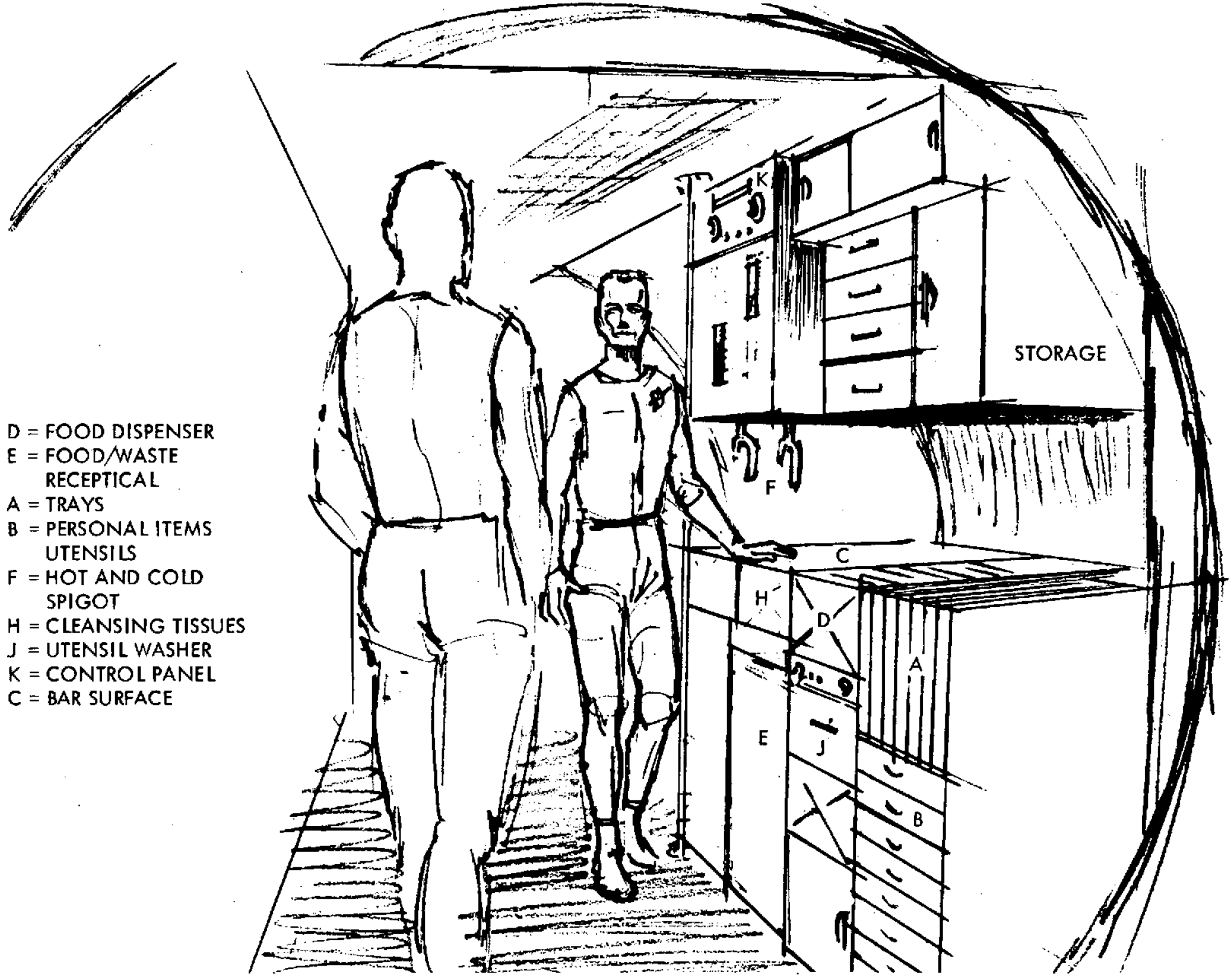


Figure 99. Food Preparation Bar

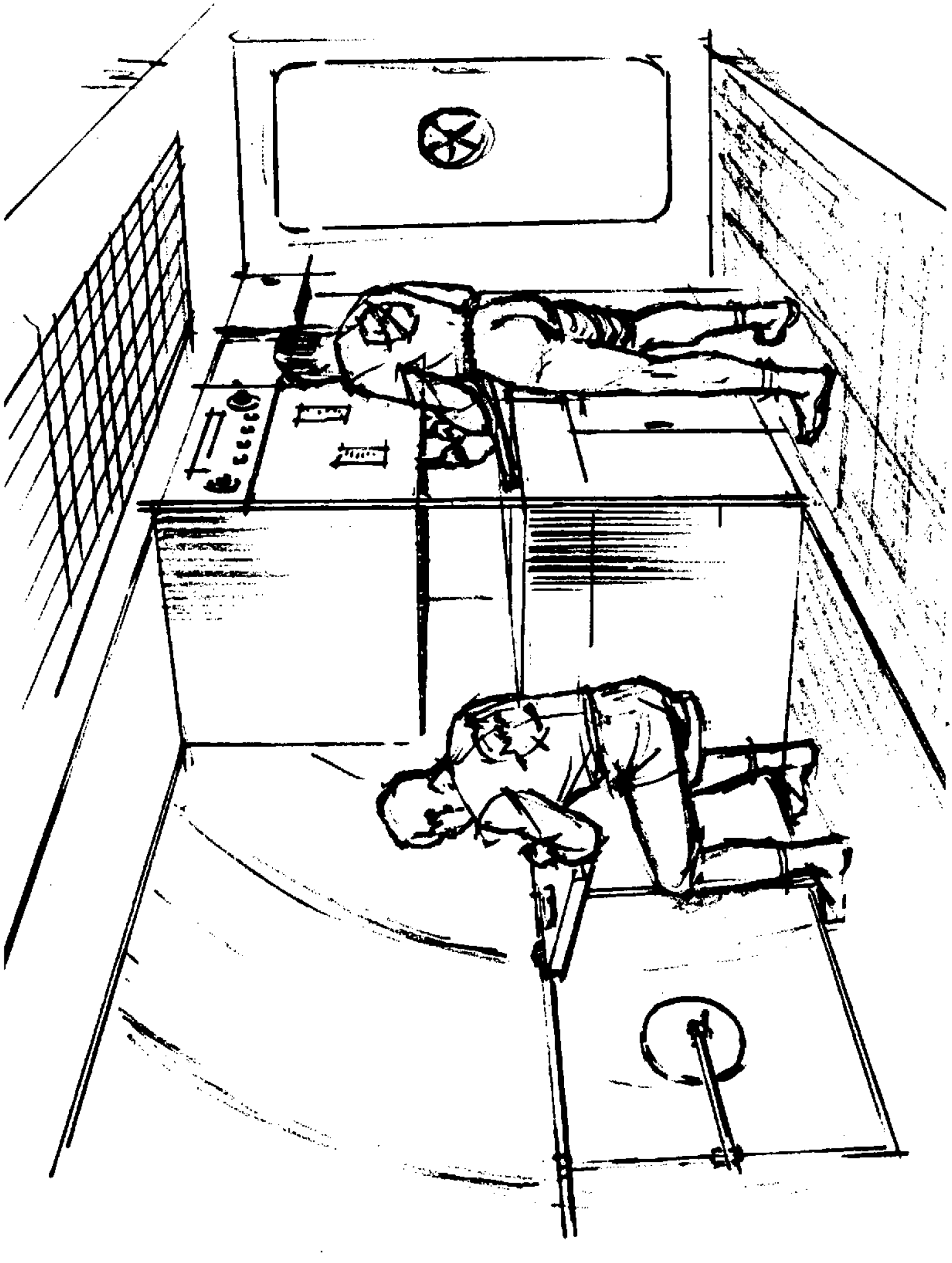


Figure 100. Eating Arrangments

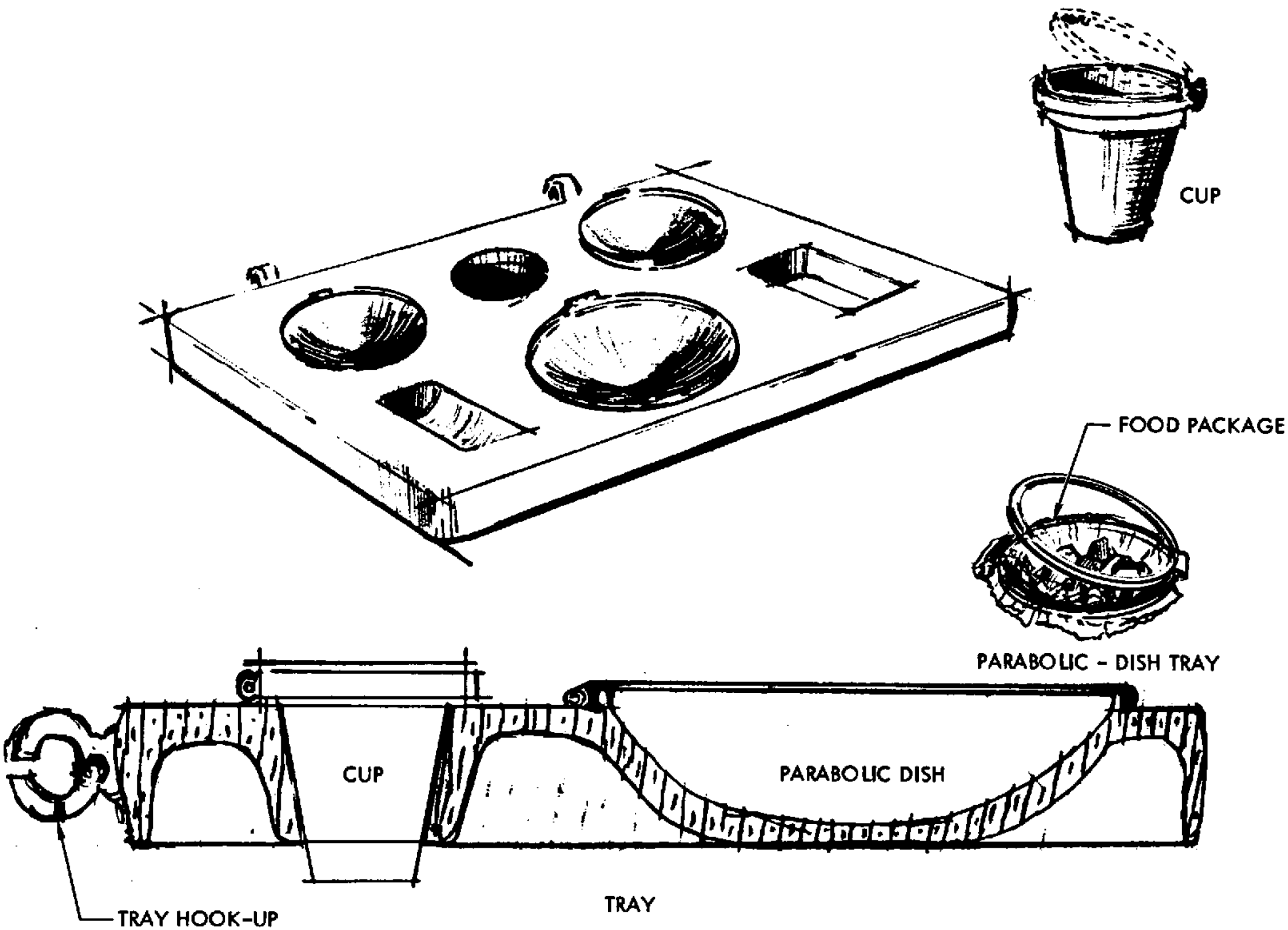


Figure 101. Food Server

mixing cup. Cleaning the tray with the cup after eating involves little more than removing the empty individual wrappings and placing them in the waste receptacle.

Weights

The approximate weight of equipment required for food preparation and storage for a 6-week period for 21 men is listed below.

Equipment	Weight (lb)
Food preparation	132.0
Food containers (132 required)	66.5
Water storage tank	18.0
Water Required:	
1. Food preparation	120.0
2. Storage (30 gal)	240.0

FOODS, NUTRITION, AND DIET

Menu selection, and food weight and volume—supplied by the Whirlpool Corporation Research Laboratories—are described in this section. Since an artificial gravity environment is available on the space station, conventional foods will, for the most part, suffice.

To keep power, food preparation, and food/waste handling requirements to a minimum, it is proposed that the menu be made up predominantly of precooked freeze-dehydrated foods, supplemented with some conventionally dehydrated foods, and room temperature stable, processed food items. Characteristically, freeze-dried foods require no refrigeration for storage, reconstitute readily in hot water, and require a minimum of instruction for satisfactory preparation. Foods will be dried to less than 2 percent moisture, thus reducing the weight by eliminating the water naturally present in foods. Freeze-dried menus can be formulated which provide high palatability and acceptable variety to satisfy man's psychological food needs. Foods after reconstitution will have a fresh, appetizing appearance.

All foods will be stored at room temperature, thereby eliminating any requirement for refrigeration. The storage life of freeze-dried foods is believed to be two years or longer. Since all foods will be precooked, there will be no requirement for an oven or other cooking equipment. The processing of freeze-dried foods result in very little loss of nutritional quality. Use of properly formulated menus of this type will insure meeting man's food requirements. Current nutrition standards or recommended dietary allowances formulated by the Food and Nutrition Board of the National Research Council serve as a guide for planning food intake.

Recommended Allowances

Without accurate reference to such factors as age, body weight, height, and physical activity of mission personnel, and the effect of vehicle environment, energy requirements cannot be reasonably estimated. Caloric allowance is, of course, best judged through maintenance of weight and health. Since personnel are considered sedentary in occupation and physical exercise is of an undisclosed nature, the energy allowance tentatively recommended is approximately 2500 calories. This figure agrees favorably with the 2400 calories ± 15 percent recommended in the 1948 allowances for a typical (standard reference), sedentary man, and it approaches the average allowance of 2466 calories of 36 individual, sedentary men living on essentially constant diets for periods of several months or more, without significant weight change. Total food calories are divided in the menus to allow approximately 15 percent for protein, 31 percent for fat, and 54 percent for carbohydrate. With the exception of protein, no recommendation is made regarding levels of intake for the normal diet. Wide variation in apportionment of these components is compatible with human health.

Allotment of 15 percent of the caloric allowance for protein is made. This amount exceeds the recommended protein allowance of approximately one gram per kilogram of body weight per day, or 70 grams for the reference man, regardless of activity. Investigations repeatedly confirm that this allowance is generous and, reduced to 50 grams, would still provide approximately a 30-percent margin above requirements to cover differences in individual need and in protein in the diet.

Approximately 31 percent of total food calories for fat seems to provide a satisfactory diet, and is used. The role of dietary fat is essential to a generally acceptable diet. It should be noted that the proposed diet supplies most of the calories from fat in the forms of visible vegetable fat. Use of nonfat, dried milk and fat removal incidental to the freeze-drying of meats permits use of an adequate amount of spread for bread items. Too, values for vegetables do not include fat which is indicated separately. Inclusion of sauces and gravies may also be related to acceptability of the diet, should significant variation in water of reconstitution be encountered among samples.

Mineral and vitamin levels in the proposed diet meet or exceed the recommended daily dietary allowances which provide a margin against an increased requirement for various stresses.

Although the diet is supplemented with baked products and dried cereals, "instant" dehydrated foods, spreads, etc., predominant use is made of freeze-dried foods. Values used to estimate nutritive levels are derived from standard reference tables and do not constitute an absolute analysis. In most cases, values presented in the menus in the following section represent averages of analytical data. They represent an effort to present values typifying year-round availability. Constituents of foods vary widely with such factors as variety, locality and season of growth, maturity, storage, and method of preparation. However, differences in initial food content may be slight in comparison with individual biological variation.

Values for precooked, freeze-dried foods are those reported for fresh, cooked foods unless such information is lacking in the tables. In these cases, raw (turkey, mushrooms) or canned (waxed beans, corn, and soup) values are used. While complete information is still not available concerning the nutritive value of freeze-dried foods, reported data on British freeze-dried foods indicates that such a comparison is valid. Losses of heat-labile constituents—one criterion of nutritive value—for reconstituted, precooked freeze-dried products are probably on the same order as those of freshly prepared samples.

Values for meat found in standard tables may be misleading, since there is generally an allowance for fat ordinarily found on various wholesale cuts. Meats subjected to freeze-drying are trimmed prior to processing, thereby altering proximate composition and energy value considerably. Since data indicate that losses of heat-labile constituents during processing of freeze-dried foods are small, fresh and frozen concentrate values are generally used for fruits and juices. Exceptions to this are canned cranberry juice and bottled grape juice values. Where applicable, enriched rather than unenriched products are used in estimating nutritive value. For reasons of stability in storage, values for margarine are based on a modified product prepared without the dry milk solids.

The following list compares the recommended daily dietary allowances with the estimated average values obtained from the sample menus prepared for a 1-week period.

	Protein (Gm)	Calcium (Mg)	Iron (Mg)	Vitamin A (I. U.)	Thiamine (Mg)	Riboflavin (Mg)	Niacin Equiv. * (Mg)	Ascorbic Acid (Mg)
R. D. A.	70.0	800	10	5000	1.30	1.80	18.0	75
Average Dietary Values	98.1	1435	14.1	9030	1.65	2.44	16.64	126

*Niacin content of foods without 14-gram niacin equivalent derived from tryptophan content of normal diet.

Approximate daily nutrient intake is tabulated and included with each menu.

Food Weight and Volume Estimates

Estimated weight of freeze-dried foods is based on 100-percent dehydration of juices and 98-percent dehydration of other foods. Prior to preparation, total food weight per man in the accompanying menus varies from 636 to 694 grams per day, resulting in an average weight of 668 grams or a little less than 1.5 pounds per man-day. Using these menus as a basis for estimating weight for a 21-man crew for a 6-week supply, food weight approaches 1300 pounds, exclusive of packaging. This weight estimate includes foods used for zero-gravity experimentation, since certain menus are readily adaptable to zero-gravity feeding. Minor modification, as elimination of the cookie or substitution of a product of softer consistency in the second day's menu, may be necessary for flexibility of use. However, other menus as that of the fourth day require no modification.

Volume of food stored varies from about 0.12 to 0.15 cubic feet per man-day or a total of approximately 116 cubic feet for food storage for 21 men for 6 weeks. Weight estimates do not include seasonings other than salt. Volume estimates do not include condiments since it is proposed that these items be stored separately and packaged in shakers for each man's use at the time of food preparation.

The seven-day sample menus (refer to Tables 12 through 18) are intended primarily as a feasibility study to show potential use of freeze-dried foods is increasing. While all freeze-dried foods used in the menus are not presently available commercially, they are all within the scope of current freeze-drying technology.

Table 12. First-Day Sample Menu

Food	As Eaten (Amt)	Weight (Gm)	Dry Weight (Gm)	Calories	Protein (Gm)	Fat (Gm)	CHO (Gm)	Fiber (Gm)	Calcium (Mg)	Phosphorus (Mg)	Iron* (Mg)	Vitamin A (I. U.)	Thiamin* (Mg)	Riboflavin* (Mg)	Niacin* (Mg)	Ascorbic Acid (Mg)
Grapefruit Juice	2/3 C.		22	75	1.0	0.2	19.4	0.10	16	26	0.6	15	0.06	0.04	0.4	69
Rice Crispies	1 C.	28	28	110	1.6	0.2	24.6	0.14	6	32	0.5		0.13	0.02	1.5	
Milk	1/4 C.; 4 t.		7	24	2.4	0.1	3.5		87	72		2	0.02	0.12	0.1	1
Sugar	2 t.	8	8	31		.	8.0		---	---	---					
Cinnamon Slices	2	110	110	355	9.4	8.6	59.2	0.22	69	114	2.0		0.26	0.17	2.4	
Margarine	2 pats	14	14	101		11.3						462				
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Minted Pineapple	1/3 C.	47	8	24	0.2	0.1	6.4	0.19	8	5	0.1	61	0.04	0.01	0.1	11
and Strawberry Cup	1/3 C.	50	6	19	0.4	0.3	4.2	0.70	14	14	0.4	30	0.02	0.04	0.2	30
Hot Open Face Sandwich:																
White Bread	1 slice	23	23	63	2.0	0.7	11.9	0.05	18	21	0.4		0.06	0.03	0.5	
Boiled Ham	2 oz.	56	28	126	14.1	7.3			6	57	1.7		0.63	0.16	3.1	
Sauce -																
Milk	1 T., 1 t.		2	6	0.6		0.9		22	---		TR	0.01	0.03	TR	TR
Mushroom Sauce	1.83 oz.	52	10	47	1.1	2.7	4.9	0.10	27	23	0.2	83	0.02	0.09	0.1	
Grated Cheddar Cheese	2 t.	5	3	20	1.3	1.6	0.1		36	25		70	TR	0.02	TR	
Rice Broth	2/3 C.	165	19	78	4.0	3.0	8.6	0.33	54	56	0.2	17	0.02	0.03	0.5	
Lime Jello	1/2 C.	120	21	78	1.9		18.2									
Brownie	1	30	30	141	1.8	8.4	16.6	0.10	11	44	0.5	226	0.04	0.04	0.2	
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Turkey	3-1/2 oz.	100	41	228	21.2	15.2			24	337	4.0	TR	0.09	0.15	8.4	
Gravy	2 T.	33	9	54	1.3	4.1	3.1		37	31	0.1		0.01	0.05	0.1	
Cranberry Sauce	1-1/2 T.	27	27	53		0.1	13.9	0.07	(2)	(2)	(0.1)	(8)	0.01	0.01	(TR)	1
Winter Squash	2/3 C.	150	20	57	2.2	0.5	13.2	2.10	29	42	0.9	7425	0.06	0.15	0.6	8
Peas &	1/4 C.	40	8	28	2.0	0.2	4.8	0.88	9	49	0.8	288	0.10	0.06	0.9	6
Celery	1/4 C.	33	3	6	0.4	0.1	1.2	0.23	17	13	0.2		0.01	0.01	0.1	2
Margarine	2 pats	14	14	101		11.3						462				
Roll	1	38	38	117	3.4	2.1	20.9	0.07	21	36	0.7		0.09	0.06	0.8	
Current Jelly	1 T.	20	20	50			13.0		(2)	(2)	(0.1)	(TR)	(TR)	(TR)	(TR)	1
Butterscotch Pudding	1/2 C.	130	27	191	4.4	5.0	32.4		144	253	0.1	195	0.05	0.21	0.2	
Coffee	1 C.; 1 t.		1	---	---	---	---		---	---	---	---	0.01	TR	0.6	
Lt. Cream	2 t.	10	3	20	0.3	2.0	0.4		10	8		83	TR	0.01	TR	TR
Sugar	1 t.	4	4	15			4.0		---	---	---					
Lemonade	1 C.	248	28	110			28.0		2	---	0.1	10	0.01	0.01	0.2	17
Misc.																
Salt	1 t.	6	6													
TOTAL			642	2522	96.4	85.7	349.1	5.28	1365	1818	14.1	9451	1.93	2.41	21.4	151

*Values for enriched products represent minimum level of enrichment.

NOTE: All listings are of composition of foods, edible portion; all measures are approximate.

Table 13. Second-Day Sample Menu

Food	As Eaten (Amt)	Weight (Gm)	Dry Weight (Gm)	Calories	Protein (Gm)	Fat (Gm)	CHO (Gm)	Fiber (Gm)	Calcium (Mg)	Phosphorus (Mg)	Iron* (Mg)	Vitamin A (I. U.)	Thiamin* (Mg)	Riboflavin* (Mg)	Niacin* (Mg)	Ascorbic Acid (Mg)
Sliced Oranges	2/3 C.	128	19	58	1.2	0.3	14.3	0.77	42	29	0.5	243	0.10	0.04	0.3	63
Scrambled Eggs	2		28	166	13.1	11.7	0.7		53	215	2.5	1047	0.10	0.30	0.1	
Whole Wheat Muffins	2	80	80	240	6.8	8.6	34.2	0.40	66	132	1.4	386	0.18	0.17	0.5	
Cherry Preserves	1 T.	20	20	56	0.1	0.1	14.2	0.12	2	2	0.1	2	TR	TR	TR	1
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Cranberry Juice	2/3 C.	165	25	92			24.0		7	---	0.3	14	0.01	0.01	0.1	3
Macaroni & Cheese	1 C.	220	97	464	17.8	24.2	43.3	0.22	420	372	1.5	990	0.22	0.44	2.0	TR
Rolls	2	76	76	235	6.8	4.2	41.9	0.15	42	73	1.4		0.18	0.11	1.7	
Margarine	2 pats	14	14	101		11.3						462				
Blueberries	2/3 C.	75	14	46	0.5	0.5	11.3	0.90	12	8	0.6	210	0.02	0.02	0.2	12
Wafer	1	5	5	22	0.3	0.7	3.7	---	1	3	0.1		TR	TR	TR	
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Haddock	3-1/2 oz.	100	35	79	18.2	0.1			23	197	0.7	---	0.05	0.08	2.4	---
with tomato sauce	3 T.	51	6	60	0.8	4.3	4.5		5	11	0.3	650	0.02	0.02	0.4	4
Parslied	1 t.	1	0	1			0.1	0.03	2**	1	0.1	91	TR	TR	TR	2
Diced Potatoes	2/3 C.	84	20	70	1.7	0.1	16.0	0.34	9	47	0.6	17	0.08	0.03	0.8	12
with cream sauce	2 T.	33	9	53	1.3	4.1	2.9		38	31		168	0.01	0.05		TR
Wax Beans	2/3 C.	83	8	18	1.2	0.2	3.9	0.42	30	19	1.4	100	0.03	0.04	0.3	4
Corn Bread Square	1	48	48	106	3.2	2.3	17.6	0.09	67	74	0.9	120	0.08	0.11	0.6	
Margarine	2 pats	14	14	101		11.3						462				
Angel Food Cake	3" sector	60	60	162	5.0	0.1	35.2		4	14	0.2		TR	0.08	0.1	
Coffee	1 C.; 1 t.		1	---	---	---	---		---	---	---	---	0.01	TR	0.6	
Lt. Cream	2 t.	10	3	20	0.3	2.0	0.4		10	8		83	TR	0.01	TR	TR
Sugar	1 t.	4	4	15			4.0		---	---	---					
Grape Juice	1 C.	254	48	170	1.0		46.2	---	25	25	0.8	---	0.10	0.13	0.5	TR
Misc.																
Salt	1 t.	6	6													
TOTAL			694	2529	98.7	86.7	346.4	3.44	1552	1817	13.8	5059	1.37	2.56	11.0	105

*Values for enriched products represent minimum level of enrichment.
**Calcium may not be available because of oxalic acid.

NOTE: All listings are of composition of foods, edible portion; all measures are approximate.

Table 14. Third-Day Sample Menu

Food	As Eaten (Amt)	Weight (Gm)	Dry Weight (Gm)	Calories	Protein (Gm)	Fat (Gm)	CHO (Gm)	Fiber (Gm)	Calcium (Mg)	Phosphorus (Mg)	Iron* (Mg)	Vitamin A (I. U.)	Thiamin* (Mg)	Riboflavin* (Mg)	Niacin* (Mg)	Ascorbic Acid (Mg)
Sl. Apricots	2/3 C.	114	19	58	1.1	0.1	14.7	0.68	18	26	0.6	3180	0.03	0.06	0.9	8
Corn Flakes	1 C.	25	25	96	2.0	0.1	21.2	0.15	3	14	0.6		0.10	0.02	0.6	
Sugar	1 T.	12	12	46			11.9		---	---	---					
Milk	1/4 C.; 4 t.		7	24	2.4	0.1	3.5		87	72		2	0.02	0.12	0.1	1
Sweet Roll	2	110	110	355	9.4	8.6	59.2	0.22	69	114	2.0		0.26	0.17	2.4	
Margarine	2 pats	14	14	101		11.3						462				
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Cr. of Mushroom Soup	1 C.		36	151	7.4	6.1	17.7	0.23	233	195	0.4	185	0.05	0.20	TR	
Sandwich:																
Cracked Wheat Bread	2 Sl.	46	46	119	3.9	1.0	23.6	0.23	38	58	0.9		0.12	0.09	1.2	
Chicken	2 oz.	57	19	99	11.9	5.4			8	118	0.9		0.05	0.10	4.7	
Parsley	1 t.	1		1			0.1	0.03	2**	1	0.1	91	TR	TR	TR	2
Mayonnaise	1 T.	15	15	58	0.2	5.5	2.1		1	5	0.1	21	TR	TR		
Pineapple Chunks	2/3 C.	92	15	48	0.4	0.2	12.6	0.37	15	10	0.3	120	0.07	0.02	0.2	22
Spice Cookie	1	15	15	71	0.8	2.9	10.2		15	11	0.5	14	0.03	0.02	0.2	
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Ham	3-1/2 oz.	100	57	284	26.8	18.8	0.5		12	193	3.4		0.63	0.25	4.9	
Raisin Sauce	3 T.	36	10	91	0.5	2.3	19.4	---	11	17	0.4	121	0.02	0.05	0.1	
Corn	2/3 C.	110	29	94	3.0	0.8	22.2	1.21	6	57	0.7	253	0.03	0.07	1.0	6
Sl. Summer Squash	2/3 C.	140	10	22	0.8	0.1	5.5	0.70	21	21	0.6	364	0.06	0.10	0.8	15
Margarine	2 pats	14	14	101		11.3						462				
Roll	1	38	38	117	3.4	2.1	20.9	0.07	21	36	0.7		0.09	0.06	0.8	
Grape Jelly	1 T.	20	20	50			13.0		(2)	(2)	(0.1)	(TR)	(TR)	(TR)	(TR)	1
Strawberry Pudding	1/2 C.	130	27	186	4.3	5.0	31.2		144	310	0.1	195	0.05	0.21	0.2	
Coffee	1 C.; 1 t.		1	---	---	---	---		---	---	---	---	0.01	TR	0.6	
Lt. Cream	2 t.	10	3	20	0.3	2.0	0.4		10	8		83	TR	0.01	TR	TR
Sugar	1 t.	4	4	15			4.0		---	---	---					
Orange & Grapefruit Juice	1 C.	248	30	110	1.0		26.0	---	20	---	0.2	270	0.16	0.02	0.8	102
Misc.																
Salt	1 t.	6	6													
TOTAL			636	2511	99.0	84.3	347.9	3.89	1430	1824	13.0	5837	1.96	2.49	19.9	161

*Values for enriched products represent minimum level of enrichment.
**Calcium may not be available because of oxalic acid.

NOTE: All listings are of composition of foods, edible portion; all measures are approximate.

Table 15. Fourth-Day Sample Menu

Food	As Eaten (Amt)	Weight (Gm)	Dry Weight (Gm)	Calories	Protein (Gm)	Fat (Gm)	CHO (Gm)	Fiber (Gm)	Calcium (Mg)	Phosphorus (Mg)	Iron* (Mg)	Vitamin A (I. U.)	Thiamin* (Mg)	Riboflavin* (Mg)	Niacin* (Mg)	Ascorbic Acid (Mg)
Tangerine Juice	2/3 C.	164	19	64	1.5	0.5	15.1	---	31	26	0.3	689	0.10	0.05	0.3	43
Farina	3/4 C.		28	100	2.8	0.1	21.8	0.10	125	29	1.5		0.13	0.07	1.0	
Milk	1/4 C., 4 t.		7	24	2.4	0.1	3.5		87	72		2	0.02	0.12	0.1	1
Sugar	2 t.	8	8	31			8.0		---	---	---					
Banana Bread	2 slices	75	75	201	3.6	5.9	34.2	0.15	12	45	0.9	410	0.09	0.09	0.8	
Margarine	2 pats	14	14	101		11.3						462				
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Hot Tomato Consomme	2/3 C.	182	15	39	1.9	0.4	7.8	0.36	(13)	(27)	(0.7)	1901	0.09	0.08	1.9	29
Casserole:																
Tuna Fish	1-1/2 oz.	42	18	83	12.1	3.4			3	147	0.9	34	0.02	0.05	0.1	
Peas	1/4 C.	40	8	28	2.0	0.2	4.8	0.88	9	49	0.8	288	0.10	0.06	0.9	6
Noodles	1/2 C.	80	15	54	1.8	0.5	10.2	0.08	3	28	0.4	24	0.11	0.05	0.8	
White Sauce	1/4 C.	66	19	107	2.6	8.1	6.1		74	61	0.1	328	0.02	0.10	0.1	TR
Rolls	2	76	76	235	6.8	4.2	41.9	0.15	42	73	1.4		0.18	0.11	1.7	
Margarine	2 pats	14	14	101		11.3						462				
Jello	1/2 C.	120	21	78	1.9		18.2									
with Blackberries	1/4 C.	36	7	21	0.4	0.4	4.5	1.11	12	12	0.3	72	0.01	0.01	0.1	8
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Roast Beef	3-1/2 oz.	100	45	231	27.0	12.8			11	208	3.4		0.07	0.20	4.8	
Gravy	2 T.	33	9	54	1.3	4.1	3.1		37	31	0.1		0.01	0.05	0.1	
Mashed Potatoes	2/3 C.	130	31	105	2.9	0.9	22.1	0.39	35	81	0.8	52	0.10	0.07	1.2	9
Diced Beets	2/3 C.	110	15	45	1.1	0.1	10.8	0.88	23	34	0.8	22	0.02	0.04	0.3	8
with lemon juice	1 t.	5	1	1	0.1		0.4		1	1			TR	TR	TR	2
Cracked Wheat Bread	1 slice	23	23	60	2.0	0.5	11.8	0.12	19	29	0.5		0.06	0.04	0.6	
Margarine	2 pats	14	14	101		11.3						462				
Apple Sauce	1/2 C.	127	28	92	0.3	0.1	25.0	0.76	5	10	0.5	38	0.03	0.01	TR	1
Spice Cake Square		70	70	245	4.1	8.2	39.1	0.07	88	84	0.4	378	0.02	0.06	0.1	
Coffee	1 C.; 1 t.		1	---	---	---	---		---	---	---	---	0.01	TR	0.6	
Lt. Cream	2 t.	10	3	20	0.3	2.0	0.4		10	8		83	TR	0.01	TR	TR
Sugar	1 t.	4	4	15			4.0		---	---	---					
Limeade	1 C.	248	27	105			27.0		2	---	0.2	TR	0.01	TR	TR	6
Misc.																
Salt	1 t.	6	6													
TOTAL			675	2537	98.3	87.0	347.8	5.05	1336	1611	14.4	5721	1.38	2.19	15.9	117

*Values for enriched products represent minimum level of enrichment.

NOTE: All listings are of composition of foods, edible portions; all measures are approximate.

Table 16. Fifth-Day Sample Menu

Food	As Eaten (Amt)	Weight (Gm)	Dry Weight (Gm)	Calories	Protein (Gm)	Fat (Gm)	CHO (Gm)	Fiber (Gm)	Calcium (Mg)	Phosphorus (Mg)	Iron* (Mg)	Vitamin A (I. U.)	Thiamin* (Mg)	Riboflavin* (Mg)	Niacin* (Mg)	Ascorbic Acid (Mg)
Stewed Prunes	2/3 C.	195	103	321	2.0	0.4	84.2	1.17	43	66	2.9	1463	0.06	0.12	1.2	2
Scrambled Eggs	2		28	166	13.1	11.7	0.7		53	215	2.5	1047	0.10	0.30	0.1	
Cinnamon Muffin	1	48	48	134	3.8	4.0	20.2	0.05	99	92	0.8	48	0.09	0.10	0.7	
Margarine	1 pat	7	7	50		5.7						231				
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Beef and Vegetable Stew	1 C.	235	55	252	12.9	19.3	16.7	0.94	31	176	2.6	2515	0.12	0.14	3.5	14
Rolls	2	76	76	235	6.8	4.2	41.9	0.15	42	73	1.4		0.18	0.11	1.7	
Apple Jelly	1 T.	20	20	50			13.0		(2)	(2)	(0.1)	(TR)	(TR)	(TR)	(TR)	1
Orange	1/3 C.	64	11	29	0.6	0.1	7.2	0.38	21	15	0.3		0.05	0.02	0.1	31
and Grapefruit Cup	1/3 C.	65	9	26	0.3	0.1	6.6	0.20	14	12	0.1	TR	0.03	0.01	0.1	26
Sugar Cookie	1	20	20	64	1.0	2.6	9.1		5	12	0.2	25	0.03	0.03	0.2	(0)
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Loin of Pork	3-1/2 oz.	100	46	244	25.9	14.8			12	265	3.4		0.91	0.27	5.6	
Stewed Tomatoes	2/3 C.	160	13	30	1.6	0.3	6.2	0.64	18	43	1.0	1680	0.10	0.05	1.1	26
Artichoke Hearts		100	18	51	3.0	0.2	11.8	1.90	39	89	1.0	170	0.06	0.03	---	4
Cracked Wheat Bread	2 slices	46	46	119	3.9	1.0	23.6	0.23	38	58	0.9		0.12	0.09	1.2	
Margarine	3 pats	21	21	151		16.9						693				
Frosted Cake Square		60	60	193	3.1	3.7	37.3	0.06	70	62	0.2	168	0.01	0.04	0.1	
Coffee	1 C.; 1 t.		1	---	---	---	---		---	---	---	---	0.01	TR	0.6	
Lt. Cream	2 t.	10	3	20	0.3	2.0	0.4		10	8		83	TR	0.10	TR	TR
Sugar	1 t.	4	4	15			4.0		---	---	---					
Cider	1 C.	249	35	125	0.2		34.3	---	15	25	12	100	0.05	0.07	TR	2
Misc.																
Salt	1 t.	6	6													
TOTAL			684	2469	97.9	87.6	345.2	5.72	1206	1769	19.0	8359	2.10	2.31	16.6	110

*Values for enriched products represent minimum level of enrichment.

NOTE: All listings are of composition of foods, edible portions; all measures are approximate.

Table 17. Sixth-Day Sample Menu

Food	As Eaten (Amt)	Weight (Gm)	Dry Weight (Gm)	Calories	Protein (Gm)	Fat (Gm)	CHO (Gm)	Fiber (Gm)	Calcium (Mg)	Phosphorus (Mg)	Iron* (Mg)	Vitamin A (I. U.)	Thiamin* (Mg)	Riboflavin* (Mg)	Niacin* (Mg)	Ascorbic Acid (Mg)
Peaches	2/3 C.	112	17	51	0.5	0.1	13.5	0.67	9	25	0.7	987	0.02	0.06	1.0	9
Puffed Rice	1 C.	14	14	55	0.8	0.1	12.3	0.07	3	16	0.3		0.06	0.01	0.8	
Milk	1/4 C.; 4 t.		7	24	2.4	0.1	3.5		87	72		2	0.02	0.12	0.1	1
Sugar	1 T.	12	12	46			11.9		---	---	---					
Sweet Rolls	2	110	110	355	9.4	8.6	59.2	0.22	69	114	2.0		0.26	0.17	2.4	
Margarine	2 pats	14	14	101		11.3						462				
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Vegetable Soup	1 C.	250	28	82	4.2	1.8	14.5	0.75	32	50	0.8	---	0.05	0.08	1.0	8
Sandwich:																
Rye Bread	2 slices	46	46	114	4.2	0.6	24.2	0.18	34	68	0.8		0.09	0.04	0.8	
Process Cheese	2 oz.	57	35	210	13.2	17.0	1.2		382	446	0.6	(740)	TR	0.24	(TR)	
Vanilla Pudding	1/2 C.	124	32	138	4.3	4.8	19.5		145	114	0.1	195	0.04	0.20	0.1	1
with Red Raspberries	1/3 C.	41	7	23	0.5	0.2	5.7	1.93	16	15	0.4	53	0.01	(0.03)	0.1	10
Molasses Cookie	1	15	15	71	0.8	2.9	10.2	---	15	11	0.5	14	0.03	0.02	0.2	
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Grapefruit Juice	2/3 C.		22	74	0.9	0.2	19.0	0.10	16	26	0.6	15	0.06	0.04	0.4	68
Baked Blue Fish	3-1/2 oz.	100	33	155	27.4	4.2			23	293	0.7	---	0.12	0.11	2.2	---
Mashed Potatoes	2/3 C.	130	31	105	2.9	0.9	22.1	0.39	35	81	0.8	52	0.10	0.07	1.2	9
Carrots	2/3 C.	97	10	29	0.6	0.5	6.2	0.78	25	25	0.6	12125	0.05	0.05	0.4	4
Corn Muffin	1	48	48	106	3.2	2.3	17.6	0.09	67	74	0.9	120	0.08	0.11	0.6	
Margarine	3 pats	21	21	151		16.9						693				
Gingerbread	2" sq.	57	57	206	2.2	9.9	26.9	TR	45	34	1.4	69	0.07	0.05	0.5	
Coffee	1 C., 1 t.		1	---	---	---	---		---	---	---	---	0.01	TR	0.6	
Lt. Cream	2 t.	10	3	20	0.3	2.0	0.4		10	8		83	TR	0.01	TR	TR
Sugar	1 t.	4	4	15			4.0		---	---	---					
Grape Juice	1 C.	254	48	170	1.0		46.2	---	25	25	0.8	---	0.10	0.13	0.5	TR
Misc.																
Salt	1 t.	6	6													
TOTAL			675	2495	98.2	85.0	346.1	5.18	1732	2053	12.4	15624	1.35	2.46	13.3	114

*Values for enriched products represent minimum level of enrichment.

NOTE: All listings are of composition of foods, edible portion; all measures are approximate.

Table 18. Seventh-Day Sample Menu

Food	As Eaten (Amt)	Weight (Gm)	Dry Weight (Gm)	Calories	Protein (Gm)	Fat (Gm)	CHO (Gm)	Fiber (Gm)	Calcium (Mg)	Phosphorus (Mg)	Iron* (Mg)	Vitamin A (I. U.)	Thiamin* (Mg)	Riboflavin* (Mg)	Niacin* (Mg)	Ascorbic Acid (Mg)
Orange Juice	2/3 C.		21	76	1.4	0.4	18.9	0.10	16	31	0.5	168	0.12	0.03	0.4	72
Oatmeal	1 C.	236	41	149	5.4	2.8	25.9	0.47	21	158	1.7		0.24	0.05	1.5	
Sugar	2 t.	8	8	31			8.0		---	---	---					
Milk	1/4 C.; 4 t.		7	24	2.4	0.1	3.5		87	72		2	0.02	0.12	0.1	1
Bran Muffins	2	70	70	212	5.8	8.6	30.6	0.20	64	154	1.4	286	0.14	0.17	1.6	
Plum Preserves	1 T.	20	20	56	0.1		14.2	0.12	2	2		2	TR	TR	TR	1
Margarine	2 pats	14	14	101		11.3						462				
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Cream of Corn Soup	1 C.		36	151	7.4	6.1	17.7	0.23	233	145	0.4	185	0.05	0.20	TR	
Salad:																
Cottage Cheese	1/3 C.	75	19	71	14.6	0.4	1.5		72	142	0.2	(15)	(0.02)	0.23	(0.1)	
Grapes	1/4 C.	40	8	26	0.3	0.1	6.7	0.20	7	8	0.2	32	0.02	0.02	0.1	2
Cherries	1/3 C.	51	10	31	0.6	0.3	7.5	0.15	9	10	0.2	317	0.03	0.03	0.2	4
Sliced Plums	1/3 C.	62	10	31	0.4	0.1	8.0	0.31	11	12	0.3	217	0.04	0.02	0.3	3
French Dressing	1 T.	15	15	59	0.1	5.3	3.0	0.05								
Rolls	2	76	76	235	6.8	4.2	41.9	0.15	42	73	1.4		0.18	0.11	1.7	
Margarine	2 pats	14	14	101		11.3						462				
Sponge Cake	2" square	40	40	116	3.2	2.0	21.8	0.08	11	44	0.6	208	0.02	0.06	0.1	
Milk	1 C.; 1/3 C.		27	97	9.7	0.3	14.0		347	278	0.2	7	0.09	0.46	0.2	2
Roast Chicken	3-1/2 oz.	100	34	174	20.9	9.4			14	207	1.6		0.08	0.17	8.3	
Gravy	2 T.	33	9	54	1.3	4.1	3.1		37	31	0.1		0.01	0.05	0.1	
Baked Sl. Sweet Potato	1	100	41	152	2.2	0.9	34.4	1.20	37	60	0.9	9510	0.10	0.06	0.8	23
Green Beans	1/2 C.	63	6	14	0.9	0.1	3.0	0.32	23	14	0.4	416	0.04	0.06	0.3	9
& Sliced Mushrooms	3	38	4	6	0.9	0.1	1.5	0.34	3	44	0.4		0.04	0.17	1.9	2
Cracked Wheat Bread	1 slice	23	23	60	2.0	0.5	11.8	0.12	19	29	0.5		0.06	0.04	0.6	
Margarine	3 pats	21	21	151		16.9						693				
Lemon Jello	1/2 C.	120	21	78	1.9		18.2									
Coffee	1 C.; 1 t.		1	---	---	---	---		---	---	---	---	0.01	TR	0.6	
Lt. Cream	2 t.	10	3	20	0.3	2.0	0.4		10	8		83	TR	0.01	TR	TR
Sugar	1 t.	4	4	15			4.0		---	---	---					
Apple Juice	1 C.	249	35	124	0.2		34.4	---	15	25	1.2	90	0.05	0.07	TR	2
Misc.																
Salt	1 t.	6	6													
TOTAL			671	2512	98.5	87.6	348.0	4.04	1427	1875	12.4	13162	1.45	2.59	18.1	123

*Values for enriched products represent minimum level of enrichment.

NOTE: All listings are of composition of foods, edible portion; all measures are approximate.

POWER SYSTEM

The primary objectives of the power system analyses were to establish requirements for and to comparatively evaluate various types of power systems—solar, chemical, radioisotope, and nuclear—for possible application to the self-deploying space station. A description of the space station power requirements and a comparative evaluation of possible sources are contained in this section.

Early studies indicated that a space station operational date of 1966 appeared feasible from all standpoints; consequently, the power system must be compatible with the state of the art in that time period. Since the space station would likely be the first manned spacecraft to have the capability for a very long mission duration, reliability of operation was one of the most important factors affecting the selection of the power system. Several other ground rules had their foundation in the overall conservative design approach taken, while others were based on agreements between S&ID and LRC. The crew size of initial space station configurations was limited to six men; however, subsequent increases in volume and weight made it possible for large numbers of crewmen to be aboard the vehicle. In order to minimize the probability of a failure in a single module jeopardizing the entire mission as well as the lives of all the crew members, each module was designed to be independent of the others—at least to the extent that it had its own environmental control and power system, and that it could be sealed by means of pressure doors or airlocks at each exit.

The use of the Saturn C-5 launch vehicle tends to minimize the importance of weight in the space station; therefore, it was concluded that a photovoltaic (solar cell) power system was the most favorable system for the early space station operations. While the cost of this type of power system appears to be relatively high, it is believed that the cost will tend to be small in comparison to the overall operational costs, and that it can be justified on the basis of power reliability and limited development requirements.

Several companies have supplied detailed information to S&ID on designs of the various major components of the power system, i.e., solar cells, mounting panels, batteries, inverters, and alternators. These reports, which are submitted separately to NASA as supplements to the S&ID report, were provided by the following companies:

Solar Cells:

Hoffman Electronics
Spectrolab

RCA
Texas Instruments

2. Batteries:

Gulton Industries
Sonotone
Yardney
Eagle-Picher

3. Power Control Equipment:

Gulton Industries
Mallory
Hamilton Standard
Electro-Optical Systems
I T & T

RECOMMENDED SYSTEM

A photovoltaic system consisting of silicon solar cells for power generation, and nickel-cadmium batteries for energy storage, is by far the most thoroughly tested and proven space power system available. At the present time, this system is the most promising power source for the space station, particularly from standpoints of availability and reliability. Many systems of this type have already proven themselves in actual satellite applications. The cost of solar cells and the battery weights, however, make this a comparatively heavy and expensive system for the high power loads of the space station. This cost is partly offset, from a total program cost standpoint, by the fact that very little development is required to bring a photovoltaic system to the reliability levels and availability dates required by the space station. As previously mentioned in this report, competitive systems such as solar dynamic ones will probably be available in the time period of interest, but will not have proven reliability figures approaching those of solar cells and batteries, nor could they be easily adapted to the concept of making each space station module self-sufficient from a power aspect.

The active life of the power supply should be limited only to battery charge-discharge cycle ability. The solar cells themselves should not degrade extensively in the low-altitude, low-inclination orbits selected for the space station since Van Allen radiation and that from solar flares will be slight in these orbits. The efficiency of the solar cells can be degraded considerably by temperature increases, but in this application the temperature can be held within the specified limits by proper solar panel design.

Damage from meteoroids and degradation should be low, and it has been accounted for in the design of the solar cell panels.

Thus, for the purposes of this study, a photovoltaic system will be presented as the primary source of electrical energy for the space station.

The specific design of the solar conversion system for the space station went through several trade-offs to determine the one system that would cope with all requirements and limitations placed upon it. The three basic solar cell arrangements have been considered: the array on the hub, the array on the modules, or a combination of both.

Solar Array on Center Hub Only

A single, multiple section, large area solar-cell array would be attached to the center portion of the vehicle. During launch, the solar array would be placed within a hollow tube which would support the Apollo vehicle and the upper ends of the six sections.

The total array power could be used to recharge the batteries located in the center hub station and to energize the distribution system through static inverters. Eight inverters and eight batteries are used to supply the loads.

A relay, located in the center hub, is used with each molecular sieve. These relays, when operated, connect the appropriate battery to the appropriate inverter and this inverter to the load. De-energizing of the relay reverses the above process, thus allowing the battery to be charged. Relays are also used in the same manner for cooking load requirements.

Rigid bus structures are used to distribute the power down the tube between the solar-cell array and the center hub station.

Although this system poses certain weight advantages, it also poses two major disadvantages: (1) most of the system's weight is concentrated in the center hub, which results in undesirable dynamic characteristics of the space station, and (2) the power distribution system is very complex in that power transmission must originate from a single module, thus reducing the degree of reliability.

Solar Array on Modules Only

Each module in the outer ring will have a silicon solar-cell array for operation during the orbital period in which the sun illuminates the vehicle. In addition, each module will have a nickel-cadmium battery to provide power during the orbital period in which the vehicle is shadowed by the earth.

In this system, power would have to be transmitted to the hub from the modules. One would transmit inverted (a-c) power, another would transmit unregulated dc, and still another would furnish regulated dc. Thus, the power load of the hub would be shared by three modules, rather than being imposed on only one.

The batteries will be switched off the line during the illumination period while being charged. The charger would provide for constant current charging. Since the load will vary, the charger contains a bus load current sensor to reduce the current charge when the load increases (the result would be a constant power drain from the solar array and hence a minimum of voltage regulation required). The switching of the batteries would be controlled by use of a sun sensor to determine the time to switch the batteries on and off the bus.

Although this system shows some promise, there are two major disadvantages: (1) the system would require transferring of power for relatively long distances, thus producing a complex system to compensate for power losses, and (2) the center hub would have to rely mainly on three modules for power distribution, thus reducing the degree of reliability.

Solar Arrays on Modules and Center Hub

This system is identical to the previously mentioned system in that separate solar-cell arrays are mounted on each outer module, but the difference lies in the fact that solar-cell arrays are also on the center hub.

This system has been selected as the one most feasible for this study, as based on five factors: (1) each module contains its own primary and secondary power source, (2) there is no transmission of power between outer modules and center hub, (3) the area of each solar array will be minimized, (4) the solar panels may be easily stored and erected, and (5) the system provides an easier method of power balance between outer modules.

This system will now be presented in detail as the electrical power system to be used on the space station.

DESIGN OF SELECTED SYSTEM

The overall system power requirements for the space station are listed in Table 19. These loads do not include the additional power required for the regeneration of oxygen in the environmental control system. Each module contains a separate independent power system. The total power allocated for each module is listed in Tables 20 through 26. For an environmental control system utilizing regeneration of oxygen by electrolysis of water. 1800 watts must be added to each module.

The basic power will be provided at 28 ± 4 volts dc. Three-phase, 400 cps, 208/120-volt a-c power is provided by means of a static inverter. A static regulator is used to provide regulated power at 28 ± 3 volts dc. In order to regenerate oxygen continuously, a separate system will have to provide 2 volts dc at 900 amperes in each module. All machinery and lights will be a-c powered. All heating elements are d-c powered. Food preparation, including heating, will be on 50 percent of the time.

Power must be available as soon as the orbit is established, for deployment of the space station. Detailed investigations have not yet been made to determine the power requirement for this function, but preliminary calculations indicate it will be less than 5 kwh. The batteries used for energy storage will be fully charged initially and, therefore, capable of producing power before deployment of the solar cells. The allowable discharge depth, and the recovery or recharge time required, will depend on what time of the year the station is launched, and at what point in the orbit the deployment operation is started. Although these factors need further study, the initial calculations indicate that the required power is well within the present power system capability, and so no additional weight penalty is imposed on the power system for the deployment operation.

Table 19. Total Power Requirements

Module	Load (watts)			D-C Bus Load (watts)		
	AC	Regulated DC	Unregulated DC	Average	Continuous	Peak
Living (1)	1240		1250	1750	1580	2810
Laboratory (2)	500		1090	1680	1510	2740
Control (3)	770	1300	1250	1660	1190	3690
Living (4)	1240		1250	1650	1480	2710
Laboratory (5)	500		990	1580	1410	2640
Living (6)	970	50	1500	1650	1480	2710
Sub-total	5220	1350	7430	9970	8650	17, 300
Center Hub	500		1450	980	810	2040
Total	5720	1350	8880	10, 950	9460	19, 340

Table 20. Power Requirements - Living Module 1

Equipment	Power (watts) AC	Regulated DC	Unregulated DC	Duty
Environmental Control:				
Blowers, etc.	470			
Molecular Sieve			170	1230 watts 20 min per 2.5 hrs
Blowers, etc., for Module 2	470			
Lighting	300			
Communications			20	
Food Preparation			100	
Total	1240		290	
Inverter Input			1560	
D-C Bus Totals:				
Average			1750	
Continuous			1580	
Peak			2810	

Table 21. Power Requirements - Laboratory Modules 2 and 5

Equipment	Power (watts) AC	Regulated DC	Unregulated DC	Duty
Environmental Control:				
Molecular Sieve			170	1230 watts 20 min per 2.5 hrs
Experiments:	200		800	
Communications			20	
Lighting	300			
Total	500		990	
Inverter Input			590	
D-C Bus Totals:				
Average			1580	
Continuous			1410	
Peak			2640	

Table 22. Power Requirements - Control Module 3

Equipment	Power (watts) AC	Regulated DC	Unregulated DC	Duty
Environmental Control:				
Blowers, etc.	470			
Molecular Sieve			170	1230 watts 20 min per 2.5 hrs
Lighting	300			
Communications		470	20	260 watts 80 min per orbit 1300 watts 20 min per orbit
Total	770	470	190	
Inverter Input			910	
Regulator Input			560	
D-C Bus Totals:				
Average			1660	
Continuous			1190	
Peak			3690	

Table 23. Power Requirements - Living Module 4

Equipment	Power (watts) AC	Regulated DC	Unregulated DC	Duty
Environmental Control:				
Blowers, etc., Module 4	470			
Blowers, etc., Module 5	470			1230 watts 20 min per 2.5 hrs
Molecular Sieve			170	
Lighting	300			
Communications			20	
Food Preparation:				
Cooking				
Total	1240			
Inverter Input			1460	
D-C Bus Totals:				
Average			1650	
Continuous			1480	
Peak			2710	

Table 24. Power Requirements - Living Module 6

Equipment	Power (watts) AC	Regulated DC	Unregulated DC	Duty
Environmental Control:				
Blowers, etc.	470			
Molecular Sieve			170	1230 watts 20 min per 2.5 hrs
Food Preparation			100	
Communications		50	20	
Stabilization and Control	200		150	
Lighting	300			
Total	970	50	440	
Regulator Input			60	
Inverter Input			1150	
D-C Bus Totals:				
Average			1650	
Continuous			1480	
Peak			2710	

Table 25. Power Requirements - Hub Section

Equipment	Power (watts) AC	Regulated DC	Unregulated DC	Duty
Environmental Control:				
Blowers, etc.	300			
Molecular Sieve			170	1230 watts 20 min per 2.5 hrs
Communications			20	
Lighting	200			
Experiments			200	
Total	500		390	
Inverter Input			590	
D-C Bus Totals:				
Average			980	
Continuous			810	
Peak			2040	

Solar Panel Design

An analysis of the power demand has been made for one laboratory module and will be discussed here. The same analysis has been made for the other five outer modules and center hub and the results tabulated in Table 26. The system considered below is non-regenerative. The following statements and assumptions were made:

1. The dark period load, to be furnished solely from the storage battery, will average 1680 watts. The net energy for the 36-minute dark period to be 1010 watt-hrs.
2. The storage battery ampere-hour efficiency to be 0.70.
3. The storage battery to have an end voltage of 24 volts after discharge. The maximum charging voltage to be 33.35 volts based on a 23 cell-in-series battery with an end-point voltage of 1.06 volts per cell and a maximum charging voltage of 1.45 volts per cell.
4. The light period average load requirement to be 1680 watts.

With the above assumptions, the following systems will drain from the battery in the dark period a total of 36 ampere-hours. The battery ampere-hour recharge requirement will be

$$\frac{36}{0.7} = 51.5 \text{ ampere-hours}$$

The solar panel must supply to the battery during the light period an average power of

$$\frac{60}{64} \text{ x } \frac{51.5}{.35} \text{ x } 33.35 = 1390 \text{ watts}$$

During the same light period, 1680 watts must also be supplied for a net load requirement of 3570 watts.

With the solar panel output requirement established at 3570 watts, the number of solar cells required can be computed from the output of a single solar cell. The output of a solar cell is given by power = area x efficiency x solar constant.

Table 26. Solar Cell Panel Characteristics

Module	Average Power (watts)	Number of Cells	Area Required (ft^2)	Area Utilized (ft^2)	Temperature Allowance (C)	Total Weight (lb)
1	1650	141,000	302	350	2.8	385
2	1680	143,000	307	350	2.5	385
3	1660	141,500	304	350	2.7	385
4	1650	141,000	302	350	2.8	385
5	1680	143,000	307	350	2.5	385
6	1650	141,000	302	350	2.8	385
Center Hub	980	83,500	180	225	2.7	248
Total	10,950	934,000	2004	2325		2558

Cells Required for One Array

3570 watts required
28.7 mw/cell for a bare cell
28.7 mw x 0.87 (13 percent assembly losses = 25 mw/cell
Available for each cell on a finished array

$$\frac{3570 \text{ w/array}}{25.0 \text{ mw/cell}} = 143{,}000 \text{ cells/array}$$

Required to meet the 3.57 kw output

The following calculations assume a panel temperature of 28 C

Minimum voltage required	33.35 volts
Diode allowance	0.80 volts
Tolerance ± 1 volt	1.00 volts
Nominal voltage	35.15 volts

Voltage/Cell at the maximum power point is 0.477 volts

0.477 volts/cell x 0.98 (2 percent voltage loss) = 0.467 volts. The voltage for one solar cell on a finished array is 4.67 volts

$$\frac{35.15 \text{ volts/series string}}{0.467 \text{ volts/cell}} = 75$$

Based on operating the solar cell at maximum power point voltage, 140 mw/cm^2 air mass one solar radiation intensity and an array operating temperature of 23 C, the array will require 75 cells per series string.

At 75 cells per series string and 143,000 cells per array the number of series strings required in parallel is as follows:

$$\frac{143{,}000 \text{ cells/array} = 1910 \text{ series strings}}{75 \text{ cells/series string}}$$

The current per cell at the maximum power point is 60.2 ma. Allowing 11 percent for assembly degradation, the final current per cell or current per series string would be

$$60.2 \times 89 = 53.5 \text{ ma}$$

The total panel current would then be

53.5 ma/series string x 1910 series strings = 102 amperes. The following has been shown. Assuming 140 mu/cm2 air mass zero solar radiation and a panel operating temperature of 28 C, 143,000 cells with an efficiency of 11-percent air mass zero average would produce 3.57 kw. This power would be at a minimum of 33.35 volts and would require 75 solar cells in series and 1910 series strings connected in parallel.

Area Required for One Array

Using a 90-percent area utilization factor and assuming a shingle-type of submodule, approximately 465 cells or 93 shingles can be placed in a 1-square-foot area. Assuming this figure, the area of the panel will be as follows:

$$\frac{143{,}000 \text{ cells/array}}{465 \text{ solar cells/ft}^2} = 307 \text{ sq ft /array}$$

as the operating temperature of the panel increases, each degree centigrade temperature rise will require an additional 1.9 volts per series string. Approximately four cells per series string will be required to provide this additional voltage. The panel used 1910 series strings or four cells x 1910 for an increase of 7640 solar cells for each degree centigrade of temperature rise. The increased area required would then be

$$\frac{7640 \text{ solar cells}}{465 \text{ solar cells/sq. ft.}} = 17 \text{ sq ft.}$$

The area available on each module is assumed to be 350 square feet. The panel at 28 C will require 307 square feet or a difference of 43 square feet. With 43 square feet available, a temperature rise of 2 C could be tolerated. Any further increase in operating temperature will require higher cell efficiency in order to maintain the required power level of 3.57 kw.

Weight of One Array

The weight given is for a panel consisting of a flat, 3/8-inch-thick honeycomb aluminum substrate with a 0.0007, 1/8-inch hex-cell core and 0.010-inch aluminum skin. This weight includes the weight of mounting inserts but no frame.

The solar cells are bonded with a flexible adhesive as 5-cell shingles. The packing density is 90 percent.

The panel will weigh approximately 1.10 pounds per square foot. The completed panel without any frame will weigh

$$1.10\ \text{lb/ft}^2 \times 350\ \text{ft}^2 = 385\ \text{lb}$$

Energy Storage Design

For the 36-minute dark period, two sources of stored energy are being considered, namely hermetically-sealed nickel-cadmium and silver-cadmium batteries. Although nickel-cadmium has low specific energy, it has proven to be the only battery thus far to exhibit thousands of charge/discharge cycles and still maintain a high percentage of its original capacity. Silver-cadmium batteries—although still in the development stage for large cells—possess rather flat characteristics with a higher specific energy than that of the nickel-cadmium battery. Thus for this study, both batteries have been considered with special emphasis on nickel-cadmium.

For this application, each outer module and the center hub will require a separate battery supply with the interface capability of being able to distribute electrical energy to adjacent modules.

The battery supplies in all of the modules must have the capability of supplying their respective loads during the dark period and completely recharge within 64 minutes of the light period with a minimum cyclical time of 5260 cycles.

It has been assumed, for the purposes of this study, that the loads will occur with the beginning of a cycle being a dark period. The batteries for each module and center hub were sized based on the one cycle requiring the most energy during the dark period.

Based on the above assumptions, a 23 cell-in-series battery has been selected with an operating range between 24 and 33 volts and the ability to accept overcharge at a current of $\epsilon/10$ for nickel-cadmium and $\epsilon/25$ for silver-cadmium (where ϵ is the battery capacity in ampere-hours).

For long charge/discharge cycles, which are required for this application, the capacity of the battery is dependent upon the depth of discharge of the battery with the continous capability of being able to recharge within the required time allotted. To date, several battery manufacturers have stated that, for a mean life of 7000 cycles, a 33-percent depth in discharge has been achieved for nickel-cadmium cells. For silver-cadmium cells, a cycle life of 2500 cycles for a depth of 20 percent has been achieved.

Each of the companies supplying information on batteries stated that within the 1964 to 1965 time period nickel-cadmium batteries would be most feasible; however, with more development, silver-cadmium batteries may also prove feasible.

The depth of discharge for 6000 discharge/charge cycles averaged between 30 and 35 percent, and the specific energies of the batteries varied between 8-12 wh/lb.

Various methods for increasing reliability were presented by all companies; and various battery sizes, weights, volumes, etc. were submitted.

With the above compiled information, the battery weights and volumes have been calculated for a 35 percent depth of discharge and 10 wh/pound specific energy and are tabulated in Table 27.

Control

The control of the varying input voltage is accomplished by use of static electronic control devices. Each module will possess its own inverter and battery charger with modules 3 and 6 also possessing a voltage regulator.

The inverters will be static devices providing 3ϕ, 400 cps, 208/120 volt a-c power. They will operate with an input swing of 24 to 33 volts dc having an average efficiency of 85 percent.

The static voltage regulators will also work on an input swing of 24 to 33 volts, providing a 23 ±1 percent d-c volt output with an efficiency of 85 percent.

System Summary

The total power system weight requirements have been tabulated and are listed in Table 28. In order to easily calculate the system parameters for increased load requirements, parametric curves have been drawn for various power levels and are shown in Figure 102. Solar panel area as a function of power has also been calculated and shown in Figure 103.

In order to regenerate oxygen, the power system for each module will require a separate solar panel and storage device. The main power system is a 28-volt system, and the regenerative system is a 2-volt system. The load requirement per module for six men is 1800 watts. The power system will be designed for one module and it will be identical for all modules and center hub.

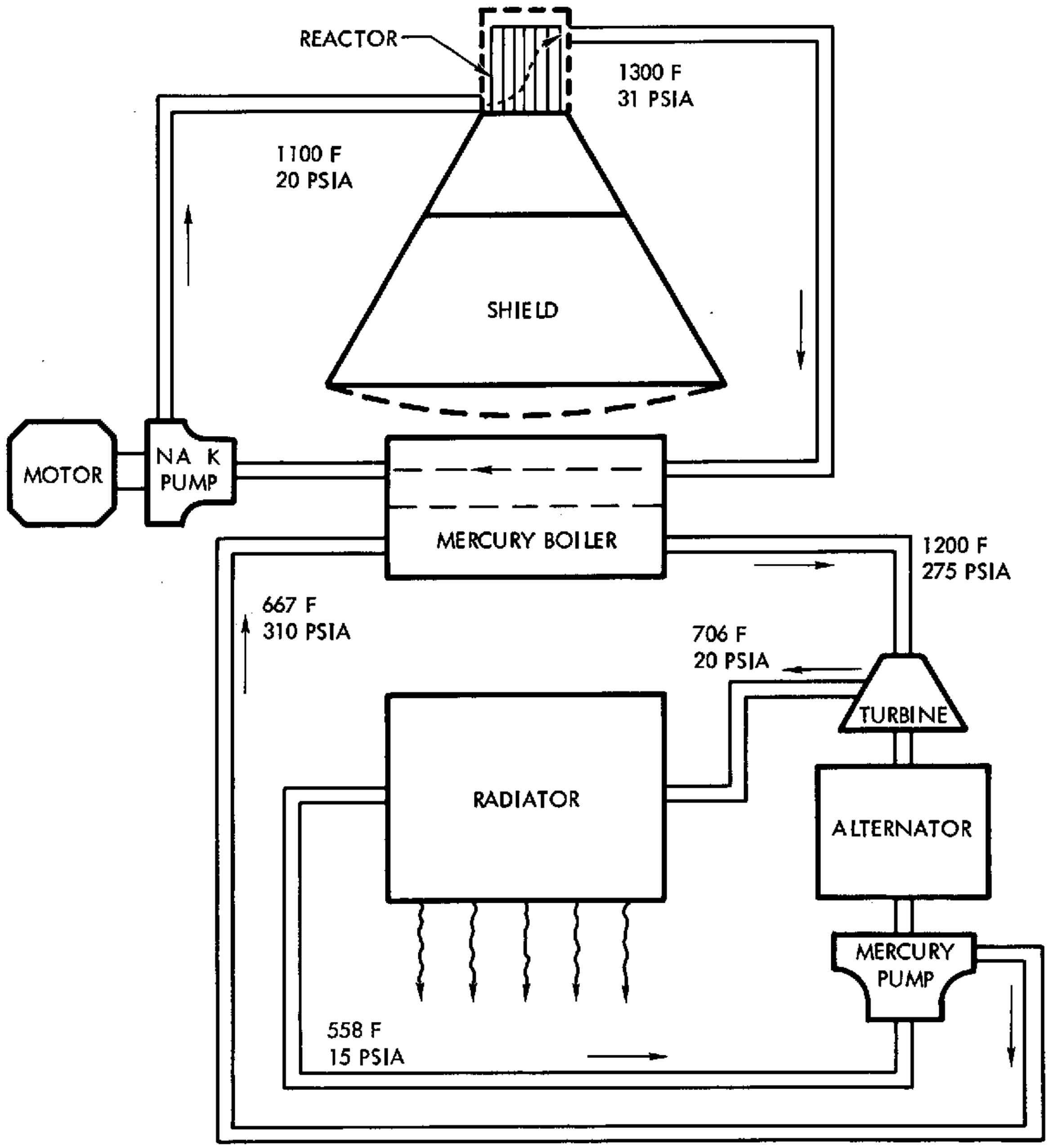


Figure 102. Nuclear Power Conversion System

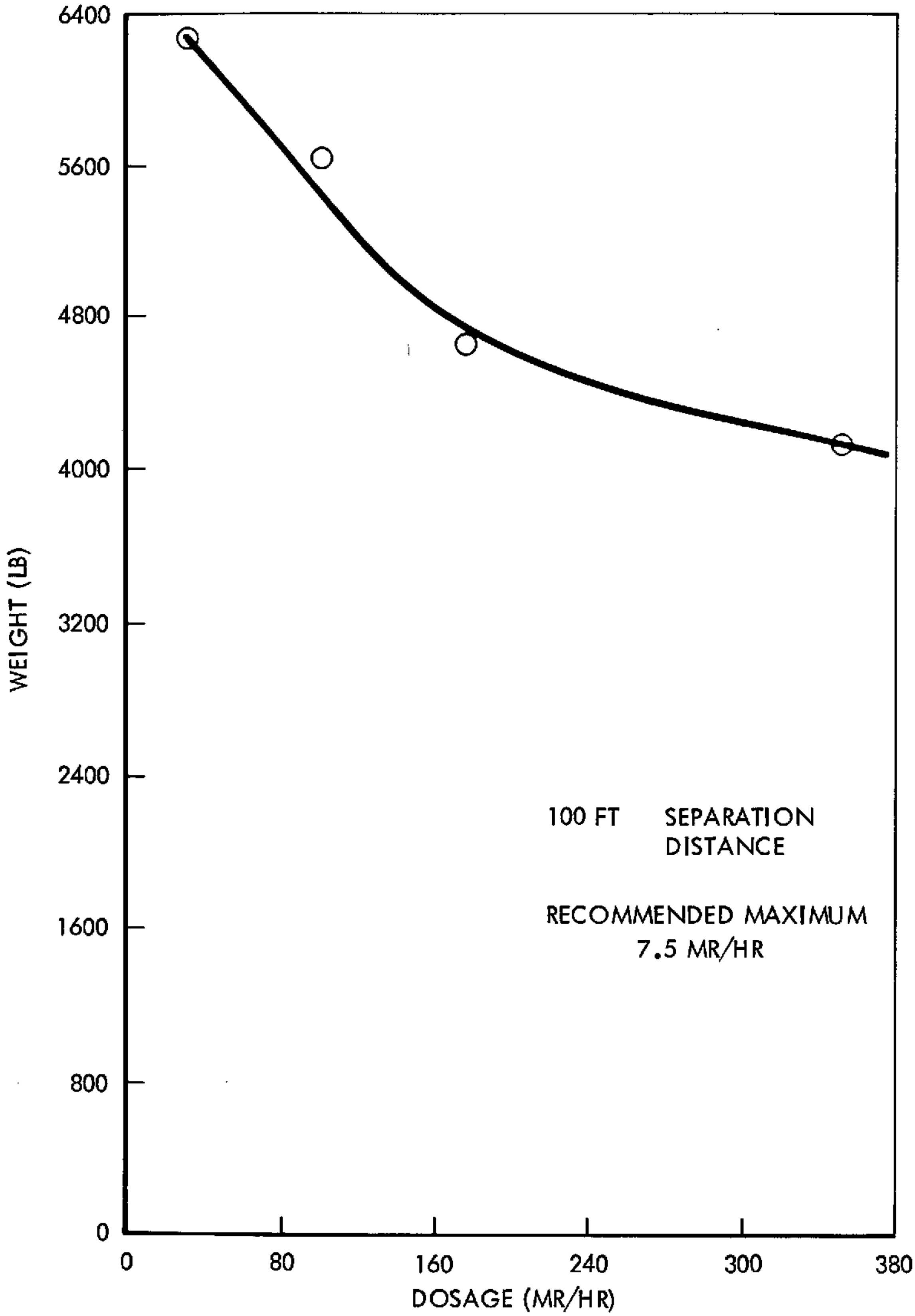


Figure 103. Shield Weight Versus Gamma Radiation Dosage (100-Foot Distance)

Table 27. Battery Characteristics

Module	Energy (kwh)	Cycle Life	Depth Discharge (percent)	Weight (lb)	Volume (ft^3)
1	3.70	6000	35	370	2.6
2	3.92	6000	35	392	2.8
3	4.20	6000	35	420	3.0
4	3.70	6000	35	370	2.6
5	3.92	6000	35	392	2.8
6	3.70	6000	35	370	2.6
Center Hub	2.58	6000	35	258	1.8
Total	25.72			2572	18.2

Table 28. Total System Weight Requirements

Module	Weight (lb)						
	Battery	Battery Charger	Solar Cell Panel	Inverter	Regulator	Misc. Wire, etc.	Total
1	370	30	385	60		50	895
2	392	35	385	30		110	952
3	420	40	385	50	10	80	985
4	370	30	385	50		50	885
5	392	35	385	30		110	952
6	370	30	385	50	5	50	890
Center Hub	258	25	248	30		50	611
Total	2572	225	2558	300	15	500	6170

NOTE: Does not include power for regeneration of oxygen

Energy during the dark period furnished by the battery

$$= \frac{1800 \times 36}{60} = 1.08 \text{ kwh}$$

Energy supplied by solar panel to recharge battery during light period

$$= \frac{1080 \times 1.4}{0.85} = 1.79 \text{ kwh}$$

Power supplied by solar panel during light period

$$P = \frac{1780 \times 60}{64} + 1800 = 3.48$$

Number of cells required per array

$$N = \frac{3480 \times 10^3}{25} = 139,200$$

Minimum voltage required	2 volts
Diode allowance	0.5 volts
Tolerance ± 0.5 volts	0.5 volts
Nominal voltage	3.0 volts

Number of cells per series string

$$N_2 = \frac{3}{0.467} = 7$$

Number of series string required in parallel

$$N_3 = \frac{139,200}{7} = 19,886$$

Area of panel

$$A_1 = \frac{139,200}{465} = 300 = \text{sq ft/array}$$

Additional area required per every 2-degree increase in panel temperature.

$$A_2 = \frac{19,886}{465} = 43 \text{ sq ft}$$

Weight of solar panel based on 23 C operating temperature.

$$W = 1.10 \times 300 = 330 \text{ lb}$$

Battery size to supply 1.08 kwh during dark period

$$\frac{1080}{0.35} = 3.10 \text{ kwh}$$

Battery weight

$$\frac{3100}{10} = 310 \text{ lb}$$

Total additional system weight for oxygen regeneration

Solar cell panels	330 lb
Battery	310 lb
Battery charges	40 lb
Wire, etc.	50 lb
Total	730 lb/module

ALTERNATE POWER SYSTEMS

Several alternate power systems were examined for possible application to the self-deploying space station. Three different solar power systems, a chemical system, and a nuclear system are discussed in the following paragraphs.

Solar Dynamic

Consideration of the Sunflower I configuration now being developed by Tapco, shows little advantage for space station use since its present output of 3 kw would have to be increased to a size which would make it unwieldy for the power requirements in question. However, several units could be place on the outer rim of the station, thus producing the required power but also providing for a much more complicated and impractical system.

The 15 kw solar mechanical engine being developed by Sundstrand shows some promise for application to the space station. Power is generated by means of rotating machinery and requires an accurately aligned solar collector with the sun.

The following list contains general preliminary design information for a single 15 kw solar mechanical engine which may prove applicable to the space station.

General Parameters:

1. Cycle fluid – Ribidium
2. Rotational speed – 24,000 rpm
3. System power output-payload – 15.0 kw

Mirror:

1. Diameter – 44.5 ft
2. Focal length – 19.26 ft
3. Rim angle – 60 deg
4. Tangential error – 1/2 deg

Heat Storage:

1. Lithium hydride weight – 136 lb
2. Sodium flouride weight – 103 lb

Radiator Design:

1. Radiator outside diameter – 99.8 in.
2. Total energy dissipation – 247,000 Btu

Total system weight – 1051.5 lb

The total system would be packaged within a cylinder 8 feet in diameter by 8 feet high.

A system of this type could be made available prior to the first flight of the space station, but the reliability of the system would be unknown. Thus, it is possible that a system of this nature would be applicable for space station use.

Solar Thermionic

Solar thermionic systems offer some promise because of their quoted possible efficiency attainment. However, these efficiencies of the order of 20 to 25 percent have not yet been demonstrated. Additionally, the stage of development is even lower than that for the fuel cell systems previously mentioned. Thus, it appears, at present, that solar thermionic systems would not apply to the space station.

Solar Thermoelectric

Solar thermoelectric systems offer some promise because they weigh less and cost less than, for example, solar cell systems; but they require approximately three times the area of solar cell systems. Thus it proves impractical to use a thermoelectric system as a power source on the space station.

Chemical Energy

Chemical power systems normally are considered in two categories — electrochemical batteries and dynamic heat engines. Batteries produce d-c power, while heat engines best produce a-c power. Thus, these systems require inverters or rectifiers to provide both a-c and d-c power.

The most appropriate comparison of chemical systems involves that of fuel requirements; for example, the best presently quoted chemically-fueled system requires approximately 1 pound of fuel per kilowatt hour based on a 10-kilowatt, continous load throughout the vehicle. Therefore, 10,000 pounds of fuel would be required for every 6 weeks of operation. This means that 85,000 pounds of fuel would be required for a 1-year mission at 10 kilowatts. Considering that this chemically-fueled power system requires mostly hydrogen, approximately 20,000 cubic feet of volume is required to store this amount of hydrogen. On the other hand, if a fuel cell is contemplated, 1/9 of this volume would be required to store the hydrogen and 1/20 of the volume would be required to store the oxygen. In either case, the tankage weight will be approximately equal to that of the fuel contained. To reduce the amount of fuel carried at the beginning of the mission, only enough fuel may be carried for 6 weeks of operation, thus reducing the prior numbers to 1/6. However, the system requires that fuel, both hydrogen and oxygen, in the case of a fuel cell, must be transported to the vehicle every 6 weeks and thence transferred from the cargo ve to the space station proper. Since the docking of incoming vehicles will be accomplished near the center of rotation a rather complicated plumbing system must be fabricated after the vehicle has been placed in orbit and deployed. Further disadvantages of chemically fueled systems, as a primary source, center on the stage of development. To obtain a low, specific fuel consumption of 1 pound per kilowatt hour, the systems under consideration would be a fuel cell or a dynamic system such as a Sundstrand Cryhocycle. Both are presently in untested stages of development as space power systems and hence do not have an established reliability such as a battery or a hydrogen peroxide turbine system. Thus, it appears that chemically fueled power systems would not apply as a primary power source for use with the space station.

Radioisotope Energy

The use of radioisotopes as an energy source for electrical power systems has been limited to something less than 100 watts. In general, thermoelectric conversion is used with a resultant efficiency of approximately 5 to 6 percent.

For a system requirement as that of the space station, no radioisotope-fueled power supply has been conceived or envisioned. The basis for the foregoing situation has been the availability and cost of the radioisotope, particularly within the time period prescribed.

Nuclear Energy

Two nuclear reactor power systems possess characteristics which appear attractive when considered for use in the manned space station. These systems are SNAP 2 and SNAP 8. Both reactors us a mercury rankine cycle to convert the thermal energy of a nuclear reactor into electrical power.

The SNAP 2 system is presently rated at 3 kilowatts with a growth potential of 6 to 10 kw. The SNAP 8 system is rated at 30 kilowatts with a growth potential of 70 kw.

The SNAP 8 system appears to be the most applicable of the two systems relative to power level. A system possessing such a high capacity results in an excellent growth potential whose experimental capabilities can include electric propulsion and high powered communications.

Since dynamic conversion of nuclear power up to 60 kilowatts (electrical) will soon be available for space applications, the following is a brief evaluation of the use of a SNAP 8 reactor system for the space station. This system is being developed by Atomics International under the ioint sponsorship of NASA and the Atomic Energy Commission. The system will have an operating life of 10,000 hours and a specific weight of 50 to 60 pounds per electrical kilowatt (not including shield). The first feasibility flight will be in 1965, with the facility for the first complete ground test now under construction.

The system will be designed for both 30 kwe and 60 kwe output. This report will consider only 30 kwe.

The reactor is designed for a constant output with a negative temperature coefficient of 200 F, for safety, and for remote start-up capability.

The system weight breakdown is as follows:

Item	Weight
Reactor	400
P.C.S. (includes radiator)	1100
P.C.S. (Start-up)	235
Rectifier and Transformer	250
Auxiliary Power	150
	2135

Component Design Data:

Turbine

Speed (rpm)	20,000
Power (hp)	61
Efficiency (%)	65
Pressure ratio	14/1

Alternator

Speed (rpm)	20,000
Frequency (cps)	1,000
Voltage (rms), ac	43.6/75.6, 3 phase, 4 wire

Power factor	0.9 to 1.0
Efficiency (%)	90
<u>Mercury Pump</u>	
Speed (rpm)	20,000
Head rise (ft)	57
Capacity (gpm)	1.1
Efficiency (%)	25
<u>Electrical Controls</u>	
Voltage regulation	±5 percent
Frequency regulation	±1 percent
<u>Na K Pump and Motor</u>	
Speed (rpm)	6,330
Head Rise (ft)	31.6
Capacity (gpm)	67
Efficiency (%)	25

The system primary loop consists of reactor, boiler, and electric motor-driven pump, using eutectic sodium-potassium alloy as the working fluid. It operates between 1100 F and 1300 F. The reactor uses zirconium-hydride fuel elements and is controlled by beryllium reflectors surrounding the core. The secondary loop consists of the boiler, turbine, radiator and pump, utilizing mercury as the working fluid. The mercury boils at 1075 F and 275 psia, and is superheated to 1200 F. The mercury vapor is then expanded through the turbine and enters the condenser-radiator at 706 F and 20 psia. There it is condensed and subcooled to 558 F at 15 psia. The mercury is also used to cool the alternator and electrical controls. The efficiency of the reactor is about 10 percent; therefore, approximately 250-kilowatt thermal must be rejected from the radiators. The radiators will function at approximately 2000 Btu/hr/ft^2 or approximately 400 square feet.

To use this system and appear feasible, it is necessary to separate the reactor from the payload by a distance of approximately 200 ft.

The shielding calculations determined an approximate weight of 3000 lb required if the reactor system could be separated from the payload by this distance. The possibility exists that the reactor could be pushed down from the center hub by a telescopic method. The shield could be designed as a shadow shield for the full 150-feet diameter of the payload. No consideration was given to the protection afforded by the space station itself. The shielding calculations were based on a flux of

$$1.65 \times 10^{12} \frac{\text{neutrons}}{\text{cm}^2\text{ sec}} \text{ and } 1.97 \times 10^{13} \frac{\text{Mev}}{\text{cm}^4\text{ sec}}$$

The shield would consist of approximately 18 inches of zirconium hydride and 48 inches of lithium hydride. The calculations were based on:

Conditions -

1. Fission spectrum source
2. Hydrogenous material

Source -

finite isotropic plane source

Assumptions -

contribution of each point of isotropic source depends only on distance between source and detector (r) and has the form $\frac{qt}{4\pi r^2}$

Dose rate infinite source

$$D(\infty) = 2\pi Sa \int_z^\infty \frac{e^{-r/\lambda}}{4\pi r^2} \, r dr$$

$$D(\infty) = \frac{Sa}{2} \int_z^\infty \frac{e^{-r/\lambda}}{r} \, dr$$

$$D(\infty) = \frac{Sa\lambda}{2z} e^{-z/\lambda}$$

Ratio D (∞) to D (a)

$$\frac{D(\infty)}{D(a)} \approx 1/2 + \alpha; \; \alpha = \frac{2\lambda^2}{a^2}\left(\frac{z}{\lambda} + 1\right)$$

$$D(a) \approx \frac{D(\infty)}{1/2 + \alpha} = \frac{\frac{Sa\lambda}{2z} \; e^{-z/\lambda}}{1/2 + \frac{2\lambda 2}{a^2}\left(\frac{z}{\lambda} + 1\right)}$$

λ = relaxation length ($\frac{1}{\epsilon}$ fast neutrons)

($\frac{1}{\mu}$ gamma rays)

z = thickness

a = radius

Sa = equivalent surface source strength $\left(\frac{\text{particles}}{\text{cm}^2 - \text{sec.}}\right)$

ϵ = neutron cross section $\frac{\text{cm}^2}{\text{cm}^3}$

μ = absorption coefficient gamma rays (cm^{-1})

Assuming isotropic distribution in the forward direction, the number of particles reaching the detector at a distance depends directly upon the solid angle dα in relation to the total solid angle

$$\left[\text{Dosage}\left(\frac{\text{particles}}{\text{cm}^2 \text{ sec}}\right) \text{Area}_1 \; (\text{cm}^2) \left(\frac{d\alpha}{2\pi}\right) \text{Area}_2{}^{-1} \; (\text{cm}^{-2})\right]$$

The normal maximum permissible exposure to external sources of radiation is considered to be 300 mirems per week and 30 neutron/cm^2 sec.

Rems = RBE x dose in rads

Probable early effects of acute radiation exposure over the whole body:

Acute Dose

0 to 25 r	- No detectable clinical effects
25 to 50 r	- Possible blood changes but no clinically detectable effects
50 to 100 r	- Nausea and fatigue and reduced vitality, depression of nearly all blood elements (recovery in nearly all cases within three to six months)
200 to 400 r	- Same as above, with immediate disability; some deaths within two to six weeks.
400	- Fatal to 50 percent.
600 r or more	- Fatal to nearly all within two weeks.

The SNAP 8 unit should be given a great deal of consideration for a space station whose availability would be no earlier than 1967. However, since the mission time required for this study is in the 1964 to 1965 time period, no more consideration will be given to nuclear power at this time.

Figure 104 shows a typical nuclear power conversion system with the shielding weight being substantiated by Figures 105, 106, and 107.

COMPARISON OF POWER SYSTEMS

Table 29 presents a summarization of the major advantages and disadvantages of the various energy sources and conversion methods considered in the study. The basic features of these systems were discussed in the preceding portions of this section; additional details are included in supplementary reports provided by S&ID associate contractors in this study.

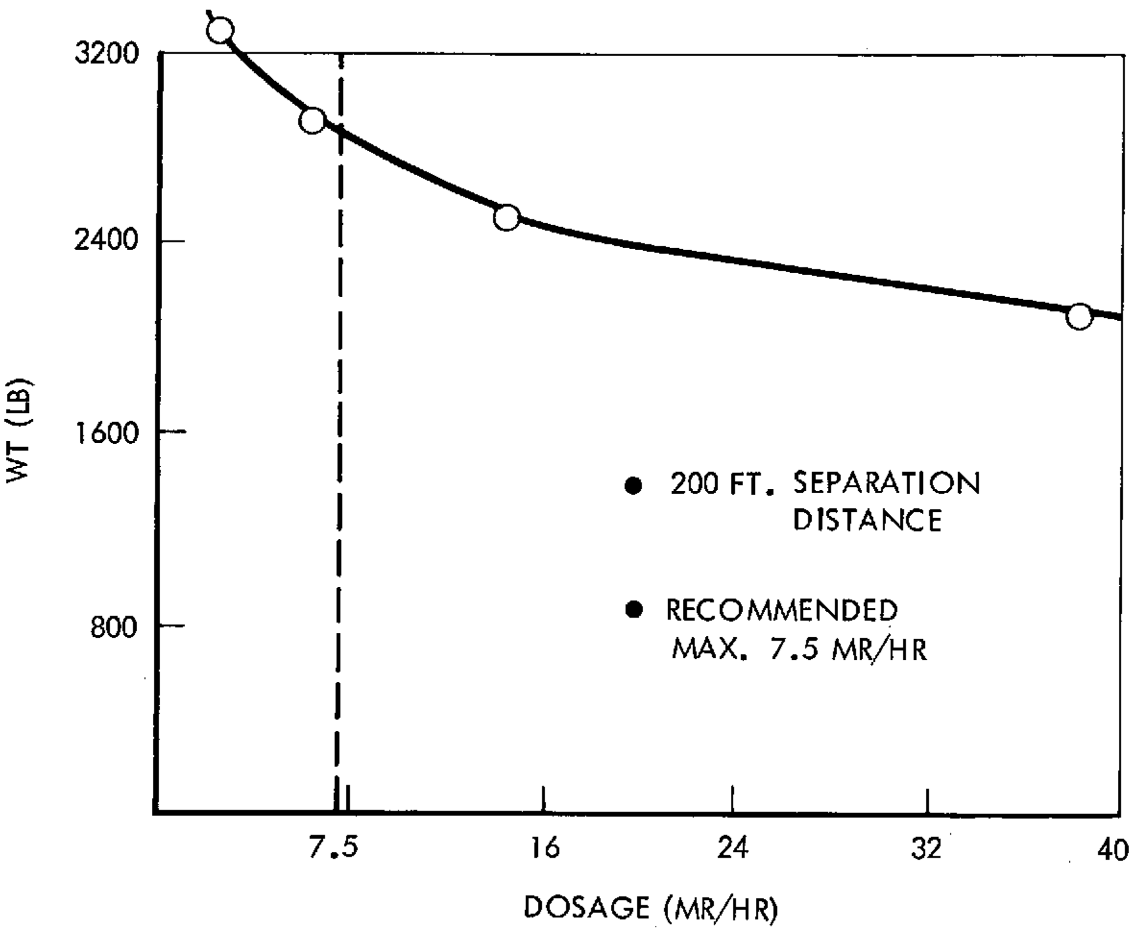


Figure 104. Shield Weight Versus Secondary Gamma Radiation Dosage (200-Foot Distance)

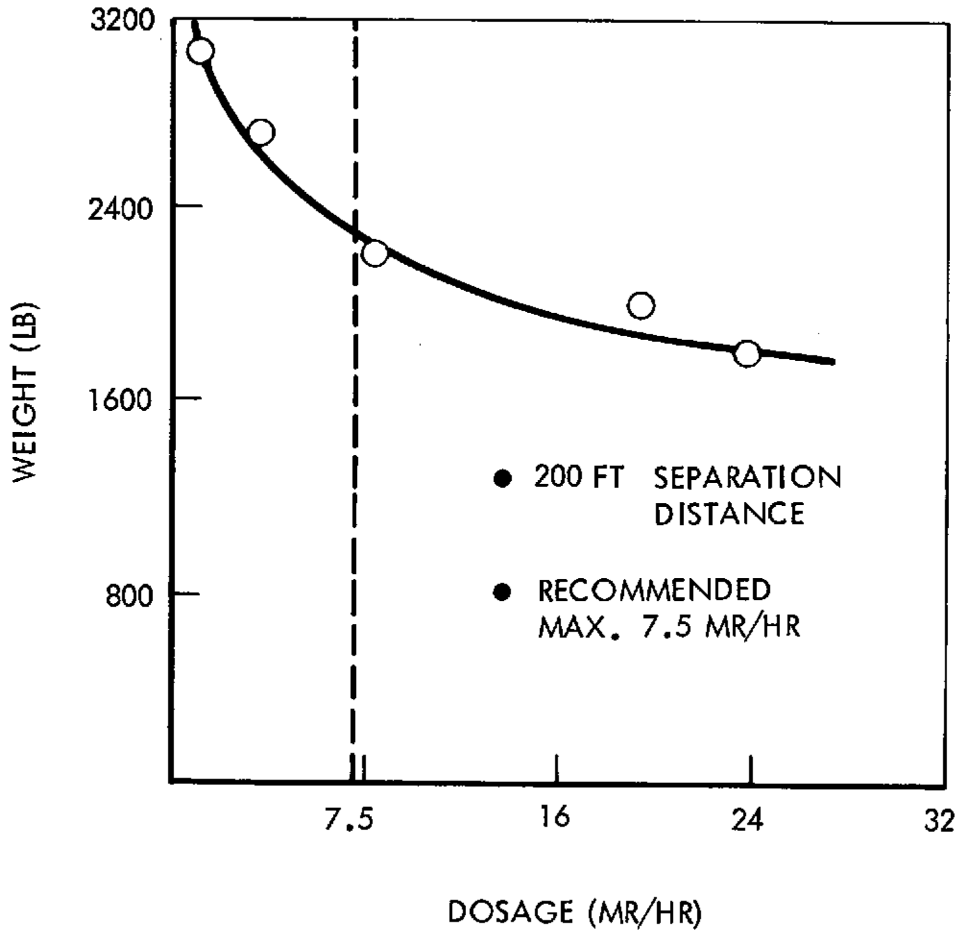


Figure 105. Shield Weight Versus Gamma Radiation Dosage (200-Foot Distance)

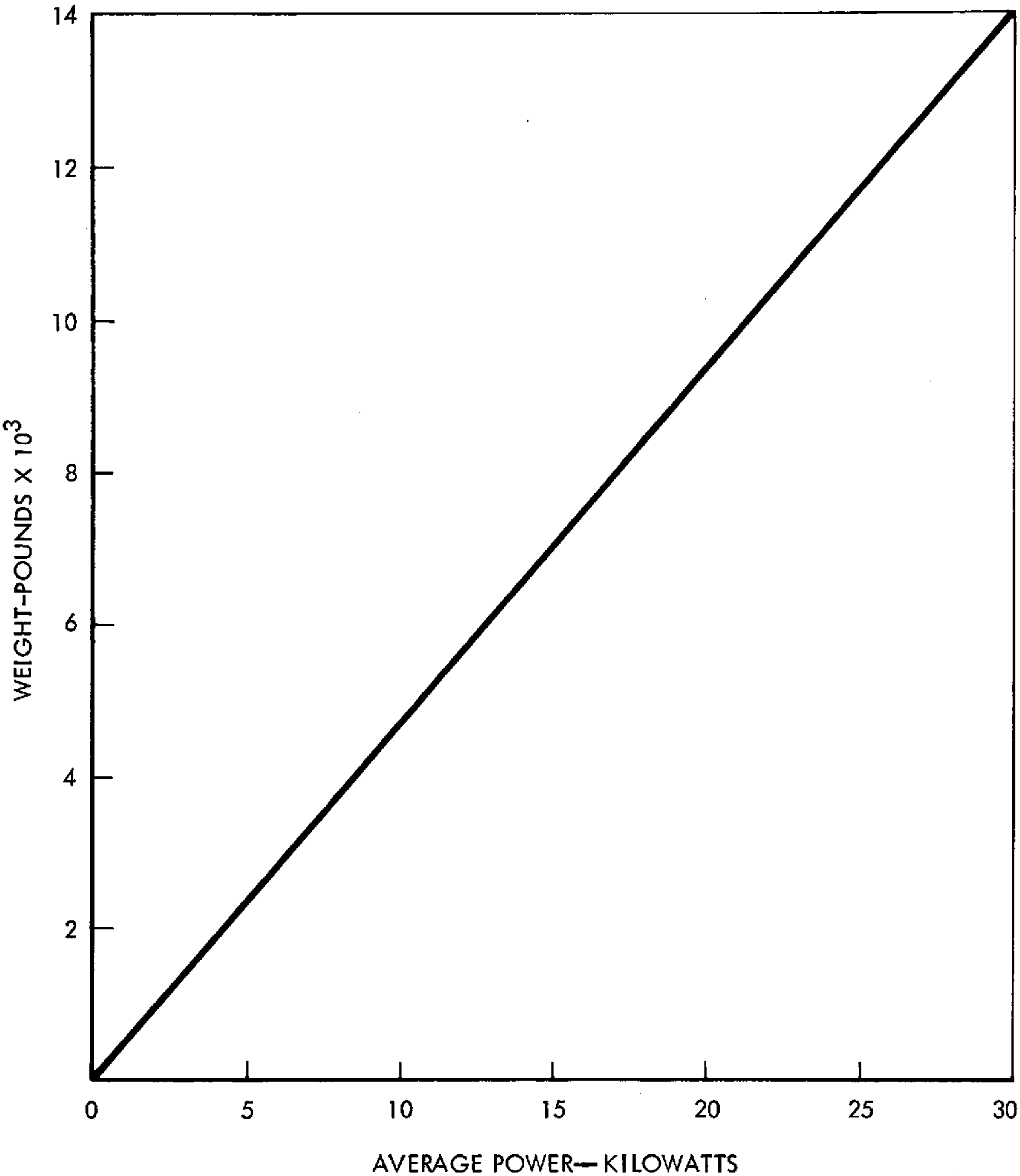


Figure 106. Total System Weight Versus Average Power

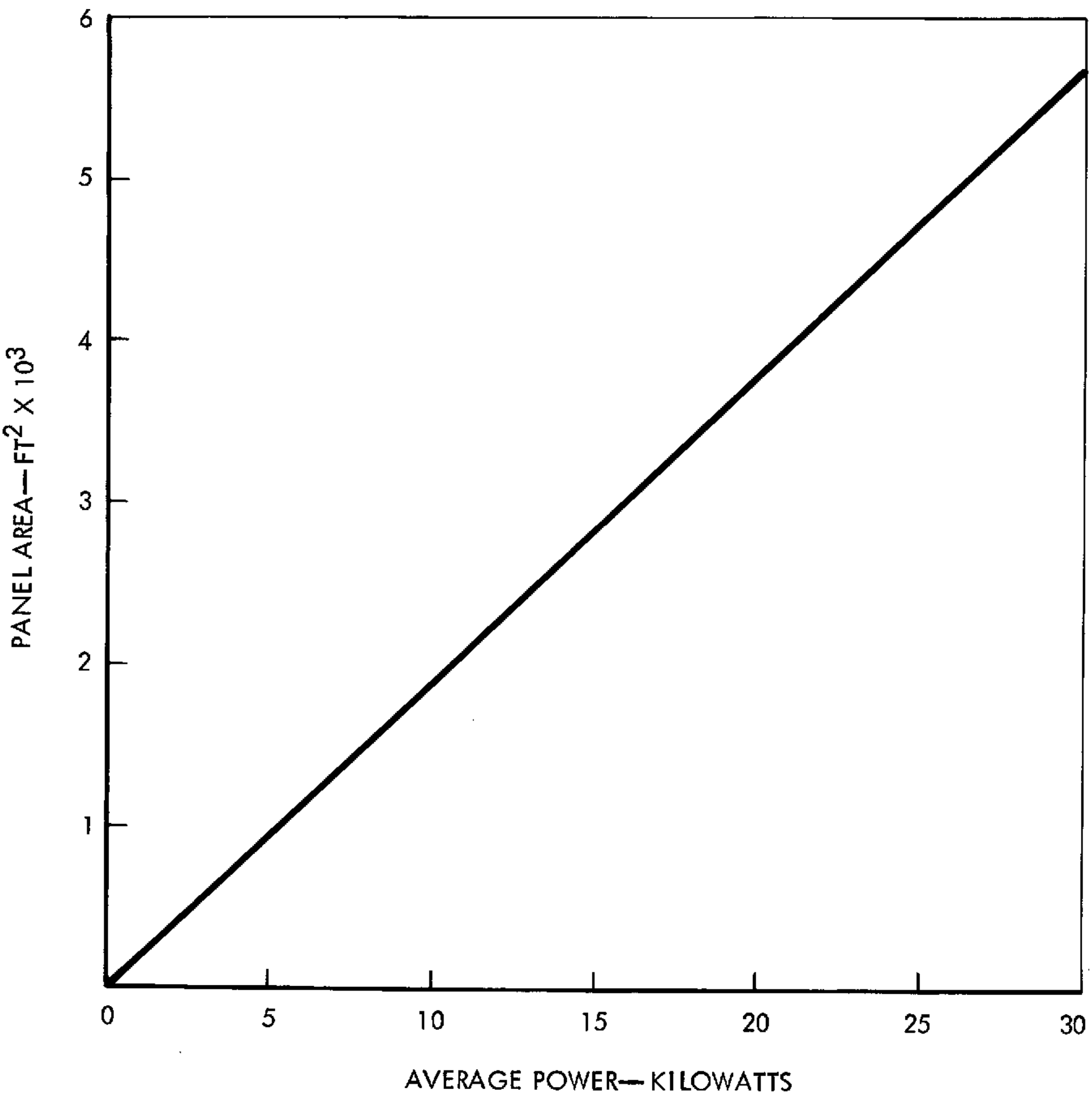


Figure 107. Total Panel Area Versus Average Power

Table 29. Comparison of Power Systems

Energy Source	Conversion and Storage Methods	Power Types	Advantages	Disadvantages	Integration Possibilities
Solar	<u>Conversion:</u> Silicon cell <u>Storage:</u> Battery	dc	1. Static conversion 2. Readily available 3. Proven highly reliable 4. Light weight 5. Small in size 6. Unlimited solar cell life	1. Low overall efficiency 2. Requires load matching 3. Requires sun orientation of collector for maximum efficiency 4. Cells are temperature, radiation, spectrum, and micrometeorite sensitive 5. Shadow period storage required 6. Increased cell area required for unoriented systems 7. High cost per watt	1. Environmental control radiator combined with power system radiator 2. Radiator on backside of collector or integrable with structure

Table 29. Comparison of Power Systems (Cont.)

Energy Source	Conversion and Storage Methods	Power Types	Advantages	Disadvantages	Integration Possibilities
Solar	Conversion: Thermoelectric Storage: Thermal	dc	1. Static conversion 2. Light weight 3. Long duration 4. High power levels 5. Relatively low cost	1. Requires precision collector 2. Requires precise sun orientation of collector 3. Requires load matching 4. Carnot cycle limited 5. Collector subject to particle damage 6. Low overall efficiency 7. Material and fabrication technique development required 8. Requires stowing until in orbit	1. Environmental control radiator combined with power system radiator 2. Radiator on backside of collector or integrable with structure
Solar	Conversion: Thermionic	dc	1. Static conversion 2. Light weight 3. Long duration 4. High power levels	1. Requires very high precision collector 2. Requires high operating temperature	1. Environmental control radiator combined with power system radiator

Table 29. Comparison of Power Systems (Cont.)

Energy Source	Conversion and Storage Methods	Power Types	Advantages	Disadvantages	Integration Possibilities
	Conversion: Thermionic (Cont.)		5. Relatively low cost 6. Small in size	3. Shadow phase storage required 4. Requires precise sun orientation of collector 5. Requires load matching 6. Low overall efficiency 7. Carnot cycle limited 8. Collector subject to particle damage 9. Material and fabrication technique development required 10. Requires stowing until in orbit	2. Radiator on backside of collector or integrable with structure
Solar	Conversion: Turbo-Generator Unit	ac	1. Good efficiency 2. High power	1. Rotating components	1. Rotating elements on the same shaft

Table 29. Comparison of Power Systems (Cont.)

Energy Source	Conversion and Storage Methods	Power Types	Advantages	Disadvantages	Integration Possibilities
	Conversion: Turbo-Generator Unit (Cont.) Storage: Thermal		3. High power density 4. Light weight	2. Complex system 3. Dual conversion method (heat-mechanical-electrical) 4. Carnot cycle limited 5. Requires precision collector 6. Requires precision sun orientation of collector 7. Collector subject to particle damage 8. Bearing life problem due to high speeds 9. Rotating components can cause interaction with attitude control system 10. Requires stowing until in orbit	2. Heat exchanger or ducts integrable with structure 3. Environmental control radiator combined with power system radiator 4. Radiator on backside of collector or integrable with structure 5. Environmental control system supplies heat for reheat cycle engine

Table 29. Comparison of Power Systems (Cont.)

Energy Source	Conversion and Storage Methods	Power Types	Advantages	Disadvantages	Integration Possibilities
	Conversion: Turbo-Generator Unit (Cont.)			11. Moving heat transfer medium (fluid-gas) 12. Zero-g effects on fluid and heat transfer 13. Fluid sealing and trans-portation problem	
Solar	Conversion: Regenerative Fuel Cell Storage: Thermal Chemical	dc	1. Very high energy density 2. High efficiency 3. Very low, SFC 4. Static except for orientation and controls	1. Requires pre-cision collector or solar cell array 2. Highly complex control system 3. Requires de-velopment 4. Zero-g problems 5. Multiple con-version 6. Low power density 7. High cost	1. Integration with chemi-cal generator ($LiH+H_2O-LiOH+H_2$) for Co_2 absorbent ($LiOH$) and H_2 supply 2. Common O_2 supply for man and fuel cell 3. Gas for fuel cell (CH_4-O_2) from man's waste 4. Environmental control radiator combined with power system radiator. 5. Radiator on backside of collector or integrable with structure

Table 29. Comparison of Power Systems (Cont.)

Energy Source	Conversion and Storage Methods	Power Types	Advantages	Disadvantages	Integration Possibilities
Chemical	Conversion: Galvanic Storage: Thermal	dc	1. Static 2. Simple 3. Reliable 4. Readily available 5. No erection 6. Minimum development 7. Low unit cost	1. Low energy 2. Cycle life limited 3. Temperature sensitive	None
Chemical	Conversion: Open Cycle Fuel Cell (H_2-O_2) Storage: Chemical	dc	1. Simple 2. Low SFC 3. Reliable 4. High energy density 5. High efficiency 6. No erection 7. Static except controls 8. Low unit cost	1. Cryogenic storage with attendant problems 2. Large fuel problems 3. Low power density 4. Requires H_2O-gas separation 5. High development cost 6. Zero-g problems	1. Integration with chemical generator ($LiH+H_2O-LiOH+H_2$) for CO_2 absorbent (LiOH) and H_2 supply 2. Common O_2 supply for man and fuel cell 3. Gas for fuel cell (CH_4-O_2) from man's waste 4. Environmental control radiator combined with power system radiator 5. Radiator on backside of collector or integrable with structure

Table 29. Comparison of Power Systems (Cont.)

Energy Source	Conversion and Storage Methods	Power Types	Advantages	Disadvantages	Integration Possibilities
Chemical	Conversion: Dynamic Heat Engine Storage: Chemical	ac	1. High power density 2. High power capability 3. Good efficiency 4. No erection 5. Available 6. Low unit cost 7. Good growth potential	1. Complex system 2. Dual conversion method (chemical-mechanical-electrical) 3. Carnot cycle limited 4. Bearing life problem due to high speeds 5. Requires H_2O-gas separation 6. High SFC 7. High development cost 8. Cryogenic storage 9. Zero-g problems 10. Gas seal problems 11. Rotating components can cause interaction with attitude control system	1. Rotating elements on same shaft 2. Heat exchanger or ducts integrable with structure 3. Environmental control radiator combined with power system radiator 4. Waste vapor for attitude control 5. Common cryogenic fuel storage tanks 6. Cryogenic fuels integrable with CO_2 removal system (freezeout) 7. Integrable with O_2 and H_2O supply for man

Table 29. Comparison of Power Systems (Cont.)

Energy Source	Conversion and Storage Methods	Power Types	Advantages	Disadvantages	Integration Possibilities
Radio-isotope	Conversion: 1) Thermoelectric 2) Thermionic Storage: Secondary Battery	dc	1. Simple system 2. Static system 3. Minimum shielding 4. No erection 5. High energy	1. Low efficiency 2. Requires radiator for excessive heat 3. High development and unit cost 4. Low power density 5. Load sensitive 6. Carnot cycle limited 7. Limited isotype availability 8. Variable heat source 9. Isotope time limited thus system must be designed for full power at end of mission life 10. Radiation hazard	1. Heat source for environmental control

Table 29. Comparison of Power Systems (Cont.)

Energy Source	Conversion and Storage Methods	Power Types	Advantages	Disadvantages	Integration Possibilities
Radio-isotope	Conversion: Regenerative Fuel Cell Storage: Chemical	dc	1. Static except for controls 2. High energy density 3. Minimum shielding 4. High efficiency 5. Very low SFC 6. No erection	1. Complex system 2. Variable heat source 3. Isotope time limited 4. Carnot cycle limited 5. Multiple conversion 6. Very high development and unit cost 7. Requires radiator for excessive heat 8. Zero-g problems 9. Limited fuel availability 10. Limited isotope availability	None
Nuclear Reactor	Conversion: Dynamic Heat Engine	ac or dc	1. High power density 2. High power capability 3. Very high energy density	1. Requires shielding 2. Dual conversion (heat-mechanical-electrical)	1. Rotating elements on same shaft 2. Heat exchanger or ducts integrable with structure

Table 29. Comparison of Power Systems (Cont.)

Energy Source	Conversion and Storage Methods	Power Types	Advantages	Disadvantages	Integration Possibilities
	Conversion: Dynamic Heat Engine (Cont.)		4. No erection 5. Good efficiency 6. High growth potential	3. Carnot cycle limited 4. Bearing problem due to high speed 5. Expensive 6. Requires intermediate heat exchanger 7. Highly complex 8. Requires radiator 9. Zero-g problem 10. Radiation hazard 11. Moving heat transfer medium	3. Environmental control radiator combined with power system radiator 4. Radiator on backside of collector or integrable with structure
Nuclear Reactor	Conversion: 1) Thermoelectric 2) Thermionic Storage: Thermal	dc	1. High energy density 2. High power density 3. Static except for controls 4. No erection	1. Requires shielding 2. Carnot cycle limited 3. Expensive 4. Requires radiator 5. Load sensitive	None

Table 29. Comparison of Power Systems (Cont.)

Energy Source	Conversion and Storage Methods	Power Types	Advantages	Disadvantages	Integration Possibilities
	Conversion: 1) Thermoelectric 2) Thermionic			6. Requires intermediate heat exchange loop 7. Low efficiency 8. Very high development and unit cost 9. Radiation hazard	
Nuclear Reactor	Conversion: Regenerative Fuel Cell Storage: Thermal	dc	1. Static except for controls 2. High power density 3. High energy density 4. High efficiency 5. No erection	1. Requires intermediate heat exchange loop 2. Complex system 3. Multiple conversion 4. Requires shielding 5. Very high development and unit cost	None

COMMUNICATION SYSTEM

The communication system, like the other space station subsystems, will be vastly more complex than any such system previously included in a space vehicle. The very large experimental payload allocation (20,000 pounds) and the presence of a capable crew will yield a quantity of experimental data that will impose severe requirements on the communication system. The system, as conceived here, includes all data-handling equipment from the point where the data is collected to the point where they are delivered to the user. By such definition, of course, all on-board data-handling equipment, as well as the ground-station processing equipment, is included in addition to the radio-frequency links.

Often during the establishment of the proposed system it has been necessary to make assumptions, since so many of the system parameters could be treated only in generalities. However, by restrictive specifications of parameters where detailed information is unavailable, it has been possible to establish a representative but stringent set of requirements. It follows that establishing the feasibility of a system fulfilling these requirements will confirm the practicability of meeting the needs of the space station. It is realized that the system chosen will not be the final system; further refinement must await a more thorough study of station needs.

Included in this section is a discussion of the requirements, the basic design philosophy, and the constraints, applicable to the space station. Following this, a system concept designed to fulfill these requirements (subject to the assumed constraints) is described, and a general development plan is forwarded that will provide the system by the prescribed time. Finally, system problem areas are enumerated and a list is presented of items to be investigated further during the remaining portion of the study.

SYSTEM REQUIREMENTS

Representative requirements for the space station communication system, as established by this study, are discussed below in terms of the following subsystems: on-board communications and monitor, television, on-board data processing, and the down and up links.

On-Board Communications and Monitoring

This system is the primary support to on-board operations for the station. Included in this category are such services as crew voice links, critical-parameters monitor, and emergency links. The function of the television monitoring, which might be considered part of this subsystem, will be discussed in the section dealing specifically with television.

Crew Communications

The crew voice system must link each of the six modules, the hub, and nearby Apollo spacecraft during routine operation. In addition, it will be necessary to communicate with men in spacesuits operating inside the station. The intermodule link used for normal operation must be capable of good voice reproduction to reduce operator fatigue and to ensure adequate intelligibility to facilitate the operational procedures.

The normal module environment will provide an ambient pressure of 1 atmosphere; thus, speaker call systems should be provided to make some of the laboratory operations less cumbersome. The links between the space station and nearby Apollo vehicles or spacesuited crewmen must also be capable of good intelligibility, as the voice communication in these instances may be supporting very critical operations.

When translating these requirements into terms of the communication system needed to fulfill them, it is found that a voice-channel band width of about four kilocycles is required. For the normal inter-module system, multiple address channels are required as well as a "call-all" capability. In addition, there should be provisions for intramodule communications, and, at any time, selected portions of the crew conversations may be recorded and introduced into the data system for either permanent record or subsequent transmission to earth.

For the station-to-spacesuit or Apollo links, the personal communication in the Apollo spacesuits will be used. The Apollo, of course is capable of operating with this link, so a common means of transmission may be used for both functions. Again these conversations may originate from anywhere within the station and must be capable of being recorded.

On-Board Monitor

The on-board monitoring system will perform the normal housekeeping chores of the station. Danger warnings will be immediately displayed, both in the danger areas and in the operations centers. Time histories of critical parameters will be maintained to anticipate failures and allow preventative maintenance. An additional function of this system will be to implement the on-board test and evaluation of failures so faulty components may be isolated and replaced.

It is estimated that there will be roughly 150 parameters per module that will require regular monitoring. It is assumed that 20 of them will require constant monitoring with the remainder being sampled periodically. All of these parameters will either be narrow-band or be amenable to simple threshold monitoring.

The failure-analysis measurements would again be narrow-band or threshold measurements; however, there would be a large number of measurements. It is anticipated that this test function would involve a computer program that could be fed to the computer whenever a specific class of failure occurred. (This test feature in no way detracts from man's function in failure analysis, but merely provides him with a strong tool that enhances his capabilities.) As was stated earlier, it is anticipated that the parameters tested in this fashion will be quite numerous; however, there seems to be little point in attempting to estimate the exact system configuration since the parameters are not well defined, and the system imposes no requirements upon the spacecraft-to-ground communications as it is a complex on-board operation.

Television

Television will find its most obvious use in the monitoring of areas or events that must remain unattended. Among the possibilities are investigations where man's presence would be harmful either to the investigation or to the man. Man's presence, for example, would be undesirable during the performance of some experiments in the zero-g laboratory. His movements would impart torques to the platform, impossible to compensate completely, which would be seen by the laboratory experiment as variations in the gravitational environment and thus affect the experiments.

Examples of the use of television for monitoring uninhabitable areas might be the monitoring of high-radiation environments (natural or induced) and the monitoring of high-temperature areas associated with plasma engines, solar-ray gathering, or materials-exposure experiments. Additionally, the television system would be used to provide extra spacecraft viewing for certain operations, and to assist in damage evaluation. The diversity and priority of these uses makes it evident that a reasonably sophisticated television system must complement the on-board equipment.

Quite obviously it would be advantageous to be able to transmit to the ground any of the above monitoring functions as well as crew behavior and condition. However, the wide band width involved makes transmission of television data very demanding upon the system. Real-time television (pictures transmitted as imaged, with no change in time scale) can be justified on the precept of public information, crew behavior observation, or major damage inspection, while the advantages of a slow-scan system (wherein the band width and recording problems are considerably reduced by use of a time-stretching technique) easily justify its use. However, the television problem becomes burdensome when a requirement is established for storage and for subsequent transmission of television having high frame rates and moderate resolutions (similar to commercial TV).

Several specific problems are associated with the last requirement. First, the recording of signals having the wide band widths presently requires the use of heavy complicated recorders with questionable reliability for space-borne application. The recorders presently available operate at maximum speeds just to record the data; thus, they cannot be speeded up to provide accelerated readout, as may be desired if large amounts of data have been recorded. The result is that transmission time must equal collection time, so the ground-station availability then establishes the constraints upon the data-gathering capacity.

The need for this form of television, however, appears to be quite real. Television formats and recording schemes therefore must be devised to facilitate this form of transmission.

The subjective nature of television information becomes apparent when translating television requirements into communication system parameters. If transmitted band width is chosen to be the parameter of interest, it is found to be a function of horizontal and vertical resolution requirements. frame rate, active scan time, and picture format. To determine definitive requirements for these parameters is very difficult. It has been found, however, that some variance in these parameters may be tolerated without significant differences being noted in the picture. This fact allows the final parameters of the television system to be dictated somewhat by compatability requirements of the overall system. Examples of such compatability constraints might be the use of common on-board cameras and simplfication of ground processing for subsequent commercial transmission. The last item implies that the general quality of the transmitted television must be at least as good as commercial standards.

When using slow-scan techniques to transmit astronomical data and whatever other pictorial information might require quality similar to high-resolution photography, it is believed that the communication system should impose no restriction upon the picture content. That is, picture quality should be limited by camera or monitor capabilities rather than by transmission parameters. Thus, this system will be determined primarily by the sensor and display components available.

On-Board Data Processing

In on-board data processing, the laboratory requirements will differ significantly from any previous system due to the quantity, complexity, and variety of data to be handled. It is expected that on-board processing will run the gamut between no processing (simple real-time transmission of raw data) to complete on-board reduction of the result of some experiments. While all laboratory data-processing functions will be performed using established techniques, the overall system complexity required to handle the varying data outputs of multiple sources will be great.

To exemplify the possible data rates and formats, this section discusses some possible experiments that might be included in the laboratory. Data rates from these experiments will vary from extremely low rates involved in the periodic monitoring of experimental operating parameters, to the extremely large rates of data collection associated with such essentially reconnaissance operations as weather, infrared, and geographical mapping. A typical scientific payload is then postulated to estimate the total amounts of data that may be expected.

With a 20,000-pound experimental payload, the laboratory is capable of many diverse and complicated experimental programs. An extremely large number of experiments has been suggested; however, these experiments fall into a relatively small number of categories having characteristic data rates. These categories are (1) space environments, (2) astronomical, (3) communication, (4) reconnaissance, and (5) materiel and component testing.

Space Environment Experiments

The radiation environment in the vicinity of earth is a composite of several different fields, most important of which are the galactic-cosmic ray, the solar, and the Van Allen radiation fields. The characteristics of these fields of particular interest are intensity (and its variation), angle of incidence, energy spectrum, and type of radiation. Experiments vary in the number of these qualities that they measure. Except for the simple instruments that merely integrate the field intensity, however, the detector itself determines intensity, direction, particle type, and energy levels to be measured. The detector also determines the energy level by its stopping power (ability to absorb the particle energy) and sensitivity. Pulse height analyzers and the coincidence schemes then give spectrum-population information.

Data rates in these experiments are primarily determined by field intensities and experimental resolution (the resolution here may refer to angle of view or to incident energy).

Data rates for such an experiment may vary over an enormous range; however, judging from current space and laboratory experiments, a reasonably sophisticated experiment might be expected to yield a data rate of 10 kilobits per second or 5×10^7 bits per orbit. Equipment for such an experiment might weigh about 20 pounds and many experiments covering the entire range of radiation would be employed.

Gravitational, magnetic, and particle-field experiments would be included in the experimental payload. The data rates from these experiments would be quite low, not significantly affecting the communication system capacity requirements.

Astronomical experimentation essentially will provide pictorial data, whether the format is a star map or a spectrograph. This pictorial information has a high data content; however, the rate of acquisition of the information will be rather slow. Much data will be transferred to earth in hard-copy form,but some information would be transferred as acquired to help direct further studies, to obtain consultation from astronomers on earth, and to provide an immediate display of interesting phenomena. If picture sections chosen for transmission were 2000 x 2000 elements encoded to 32 levels,and an average of 15 such sections per orbit required transmission to earth, the average collection per orbit would be 3×10^8 bits. Since most of this information will be on a hard copy in the laboratory, no provision need be made to accomodate large peak loads, as transmission times may be averaged.

Mapping

In this category would be listed the various efforts to improve geographic and weather mapping techniques. Radar, infrared, and optical sensors may be used; and the information rate may vary greatly with operating procedures, the required resolutions, and the number of sensors. From various studies concerning such investigations, it appears that experiments of this type may be expected to average the equivalent of nearly 2×10^9 bits per orbit; peak loads of at least twice this may be expected.

Communications Experiments

Many experiments will be conducted to refine the art of space communications. Investigation will be undertaken in the use of all parts of the spectrum and various propagation modes. Special techniques will be developed for the particular applications of space-to-space communications, and an investigation will be made of the various sources of interference. This last investigation will require the monitoring of earth transmissions to determine their effect as interferences. While most of these experiments will yield low data rates, this interference investigation may be expected to yield vast quantities of information. It is not inconceivable that data rates here may approach 2×10^{10} bits per orbit.

Materials and Components Testing

A great deal of testing will be done in this field; however, the expected data rates are low and are not expected to affect the system design.

Typical Experimental Payload

Table 30 shows a typical experimental payload that might be operative on the laboratory during some phase of the mission. The representation is, of course, a gross simplification of the picture, but it serves to show the order of magnitude of the data to be handled by the system.

Table 30. Data Transmission Rates From Typical Experimental Payload

Activity	Weight (lb)	Format	Rate/Sensor	Bits/Orbit
Monitor and Housekeeping		Digital	10 K bits	2.1×10^7
TV:				
1. Real-time		Analog	2 mc	1.5×10^{10}
2. Stored		Analog	2 mc	1.5×10^{10}
Environmental Experiments	1350	Digital	10 K bits	5×10^8
Astronomical:				
1. Observatory	1200	a/d		3×10^8
2. Radio	450	a/d		3×10^8
Mapping	2000	a/d	200 Kc	1.5×10^{10}
Communication Experiments:	1500			2×10^{10}
1. Guidance	250			
2. Materials	400			
3. Structures	400			
4. Environmental Control	120	a/d	1 K bit/sec	2×10^7
5. Life Sciences	8000			
6. Power	250			
	15,920			

CONSTRAINTS

Although the communication system will suffer from the same constraints characteristic of all space systems, the gross difference in system size causes a major shift in the importance of the constraints. Whereas existing systems have, as their primary design parameters, the reduction of system size weight and power requirements, the primary station parameters will be the provision for the complicated operational requirements of the station. The following sections discuss in detail the constraints under which the communication system for the station must operate.

Spacecraft Physical and Operational Constraints

Although exact orbital parameters remain to be fixed, it is currently assumed that an orbit having very little eccentricity will be established at a nominal altitude of 300 nautical miles. The orbital inclination probably will be less than 40 degrees, with the first launch most probably the 28-degree, minimum-thrust launch from Cape Canaveral. This is a rather low orbit, and, while it imposes few demands because of the communication distance, a severe limitation is placed upon the line-of-sight availability of particular points on earth. The effects of this restriction are discussed more in the presentation of the ground-station configuration.

The laboratory is to be spin-stabilized at a spin rate of 3 to 4 rpm. An attitude-control system orients the spin axis toward the sun and damps out any perturbations. This solar-oriented operation requires that the spacecraft have the capability of directing missions in any given direction. While full coverage is not necessary on any one pass, all pointing directions will be required during the course of the mission.

The figures presented in "Configuration Analysis" illustrate the vehicle configuration. The skin of the space station is metallic, and each module is an independent unit. Internal logistics will then be determined by the airlock configurations and the sizing of the passageways. One particularly inconvenient feature is the docking facility, in that it forms an obstruction that may disrupt antenna patterns. Antenna systems must therefore be carefully considered to establish the required effectiveness.

Ground Systems

One of the most important factors affecting the design of a satellite communication system is the determination of the ground support system. The ground-station configuration affects the frequency and duration of periods when communication is possible, and thus serves to establish such system parameters as required rate of transmission, on-board storage requirements, and operational procedures.

Two basic operational premises of the space station study contribute directly to the selection of a ground-station configuration. Most important, the communication and tracking functions for the launch and original-orbit determination are to be performed by the Apollo GOSS network. Thus, the stations that directly support the space stations are required only to fulfill normal data-transfer functions. Secondly, the concept of the space station as a self-commanding, independent vehicle eliminates the need for continuous communication between the station and earth. It should be remembered here that Apollo emergency routines will be followed, and that in the event of some emergency the Apollo network would be used. However, some system other than the Apollo GOSS is desired for the normal communications function, not only from an operational viewpoint (the Apollo network could not give the space station primary service because of obligations to other operations) but because of the significant difference between the Apollo and space station data-transfer requirements.

The problem of ground communication associated with the space station becomes quite complex. The amount of data that must be transferred precludes the use of present types of ground-communication networks from points overseas within the necessary time frame. Further, while communication satellites may be used to facilitate system growth for subsequent missions, the problems expected to be associated with these satellites during the early operational period of the station make it unadvisable to base the primary communication links on their use at that time. There remain, then, the alternatives of transporting tapes collected by stations scattered about the globe, or of communicating with the data reduction center (assumed to be Manned Spacecraft Center, Houston, Texas) either directly or via land-based microwave links from other stations within the Americas.

The first alternative is unattractive because of the cost and time lag associated with the transportation of the recorded information. The second alternative greatly limits the time during which the data may be transferred. This arrangement imposes severe storage and transmission band width penalties upon the space station; however, these penalties do not prove to be prohibitive, so the ground-station configuration forming the basis of this study employs this second concept. This selection was based upon the improved data-reduction characteristics that result, and the fact that the added spaceborne capability required assures the feasibility of a system that can operate under less demanding requirements.

This system may be implemented in several ways, but the following examples will serve to demonstrate the workability of the concept.

Stations will be set up on the east and west coasts of the United States. These two facilities will be linked by microwave to the command center. The most advantageous location for each station would be determined by the orbital inclination. Since, during the course of the mission, the projection of the orbital path will form a band between the latitudes of the orbital inclination, only the latitude and the relative position of the two stations is of importance. Some concession to economic factors is to be expected, and thus it is recommended that the stations be situated on the sites of existing stations. (Examples might be the Goddard and Goldstone sites.) The new facilities might then make use of the existing services and properties. These stations would receive transmissions from the space station and relay them by direct microwave to the command center, where the transmissions would be displayed, recorded, and be available for reduction in virtual real-time.

Another form of operation (that suggests the sites mentioned) is to perform the major data reduction at the collection site and relay (real-time) only that information of immediate interest. Either operation would provide an efficient and relatively inexpensive (in comparison with other approaches) means of handling the data.

It is certainly not assumed that these two stations would be the only ones used. Many experiments, for example, would be coordinated with various stations throughout the world. Further, it would often be desirable to transmit data directly to these other stations for support of their various functions. (Ground data transfer in this instance would be limited, and would use existing links) nevertheless the use of a two-station configuration for the main data terminals serves to establish the system configuration.

Environmental Conditions

The environmental conditions to which the system will be exposed are presented in the following listing. Normally, required environments can be provided during operation; however, the system must be capable of withstanding the environments of storage, launch, and free space without jeopardizing their operational reliability.

Environmental Conditions:

High Temperature	-	150 F operating; 165 F non-operating
Low Temperature	-	0 F operating; 65 F nonoperating
Normal Temperature	-	60 to 90 F

Thermal Shock	-	Not yet determined
Low Pressure	-	10^{-6} dynes/cm^2
High Pressure	-	25 psia
Ambient Pressure Range Above	-	Any pressure between the pressures shown
Salt Spray/Fog	-	As near ocean beaches
Radiation	-	5 REM max. normal, 100 REM max, emergency
Fungus	-	As in tropical areas
Sunshine	-	None
Rain and Snow	-	None
Sand and Dust	-	As in desert and ocean beach areas
Humidity	-	To 100 percent
Acoustics	-	To 136 db at 37 to 9600 cps
Acceleration	-	20 g in any direction
Vibration (operating)	-	±2.3g, 20 to 20 cps;
		0.018 in d. a., 50 to 112 cps;
		11.4g, 112 to 2000 cps
Vibration (nonoperating)	-	±3.5g, 5, to 20 cps;
		±1.5g, 50 to 300 cps
Impact Shock	-	20 g for 11 milliseconds in any direction
		40 g (nonoperating)

BASES OF SYSTEM DESIGN

This section asserts the basic communication philosophy upon which the system will be hypothesized. The fundamental precept has been to establish feasibility by designing a system adequately handling the communications requirements of the space station while providing flexibility, reliability, and capacity. These features have been incorporated while using components with realistic development schedules.

Capability

The fundamental goal of this study was to establish the feasibility of constructing a large self-deploying space station. It became evident early in the study that the station was feasible and could be made operational by early 1966, implying a corresponding schedule for the communications system. It is extremely difficult to establish the exact requirements of such a system, however, due to lack of precise knowledge of experiments to be performed, of the ground station environment in which the system will operate, and of the operational procedures that will be followed. The payload allotment of 20,000 pounds for experimental equipment is far beyond what has formerly been possible, and both support requirements and operational requirements for such a payload are yet to be developed. The system design therefore has been based upon obtaining the highest system capabilities possible while operating under reasonably restricted operating conditions. This rule permitted the hypothesis of a set of system requirements, and the system was then designed to provide optimum service for these requirements. It is realized that further investigation may prove the system assumptions to be somewhat in error; however, establishing the feasibility of a system meeting the assumed requirements will prove that an adequate communications system can be provided for the space station within the required time scale.

Relation to Apollo

Because it forms the most efficient means of transportation to and from the space station that will be available, the Apollo spacecraft has been postulated for the personnel transport and resupply operations. An Apollo spacecraft with a three-man crew aboard is initially launched with the space station, allowing the launch and orbital-determination tracking to be performed by the Apollo ground complex. Communications during this pre-erection interval are also to be performed with Apollo systems. Further, all de-orbit procedures — whether routine or emergency — will make use of established Apollo systems and procedures. Apollo equipment, additionally, will be used aboard the space station wherever compatibility requirements or applicability make it advisable.

Flexibility

Flexibility is of prime importance in this system because the only thing definitely known about data formats and rates is that they will vary over a wide range. Different experiments will be performed during the laboratory's mission, and there will even be variations in data rates from single experiments as measured quantities change and as redundant information is rejected. Flexibility will be inherent in the system because of the use of modular system design and the inclusion of alternate operational modes. These alternate modes may be programmed either by the general-purpose computer to be installed in the space station, or by the system operators.

Reliability

This system parameter is of primary importance whenever system down-time must be held to a minimum, or whenever faulty component replacement is inconvenient or impractical. During the early stages of the NASA space investigation, considerable emphasis was placed upon component reliability because of the obvious disadvantages of replacement techniques. With the system concept associated with this station, however, reliability may be more economically achieved by using a well-conceived spares system than by attempting to provide each component with what may be termed ultra-reliability. A well-conceived spares system may be considered to include a system design concept wherein the different sub-systems are made up of basic building blocks, as many of which are identical as possible. Using this precept, system reliabilities can be greatly increased by the inclusion of a moderate number of spares.

Crew Function

It will be man's function ultimately to provide the required system flexibility. The system functions will vary so widely that it will become impractical to attempt complete automatic programming of the system. The operator will use program patch panels, system mode selection, and component replacement to adapt the system to serve, most efficiently, a particular purpose. In addition, the operator will make use of a system test program to predict or isolate failures. In this manner, the reliability-through-selected-spares concept may be implemented, and system reliability may most reasonably be brought to the required level.

The experimenter's selection and control of the experiments will have a strong effect upon the system, since alternate experiment configurations will require various data-handling procedures. The operator may require on-board data reduction for verification of data validity, editing capability for the elimination of redundancy or errata, data compacting, permanent record on-board storage, or special computer programs for experiment control. All of these forms of data handling will be selected and controlled by the laboratory personnel.

Development

With a 1966 operational target date for the space station, a communication system must be fully developed by mid-1965 at the latest. Full-scale system development is not anticipated to begin until sometime in 1963, so the total system development time will be between 24 and 30 months. With this in mind, there will be a great emphasis upon the application of currently available components and techniques. However, the anticipated reliability and system integration requirements would seem to preclude the extensive use of systems other than those manufactured to the precise specifications determined by station requirements. Hence, the intent of the "use of currently available components and techniques" statement is to limit the development period required for the equipment manufactured specifically for the station rather than to imply use of standard off-the-shelf items.

SYSTEM DESCRIPTION

A communication system that complies with the foregoing parameters has been established. General examination of this system indicated that no insurmountable problem appears to exist. Although the complexity of the system tended to dictate the need for a more detailed study of the system to ensure development by mid-1965, the limited funds available did not permit such effort. Hence, several companies specializing in the field of communications were given a specification of this system in order to conduct a more detailed study. The following five companies conducted studies of the space station communications system:

1. Airborne Instruments Laboratory
2. Bendix Systems Division
3. Hoffman Electronics Corp., Military Systems Division
4. Motorola, Inc., Military Electronics Division
5. Sylvania Electronic Systems

These studies, which have been forwarded to NASA as supplementary data, are quite comprehensive. In each case, the conclusion reached was that the system was not only feasible, but in fact presented no severe design problems.

The overall block diagram of the proposed system is shown in Figure 108, and the system physical characteristics are tabulated in Table 31. The following specific subsystems are discussed:

1. On-board communication and monitor

2. Television

3. On-board data processing

4. Transmission links

5. Ground station

On-Board Crew Communications and Monitor

This system provides the operational support functions required in the space station. Since each module is divided into five compartments and is a self-sustaining unit, the on-board communication and monitor functions are of prime importance.

The crew communications system will take two separate forms. For normal operation, the space station will be equipped with a hard-line intercommunication network having six separate channels. Each channel will have multiple-access capabilities and be capable of operating either with headsets or with speakers. Final channel routing and distribution of access points must await a more thorough development of the station operations concept; however, each separate compartment will be accessible from any other, and a general call system will be incorporated.

In addition to the above system, each isolated section of the space station will be provided with a transceiver that links an operator in a spacesuit to the normal intercom system. This system will be supplied with an emergency power source so communications can be maintained in the event of a local power failure. This same r-f link will be used to communicate with spacesuited personnel outside the station and with nearby Apollo spacecraft. In this manner, a general distribution can be made of all conversations involved in critical operations. These transceivers will be compatible with the established Apollo personnel communications units; they will weigh roughly 2 pounds each and will consume negligible power.

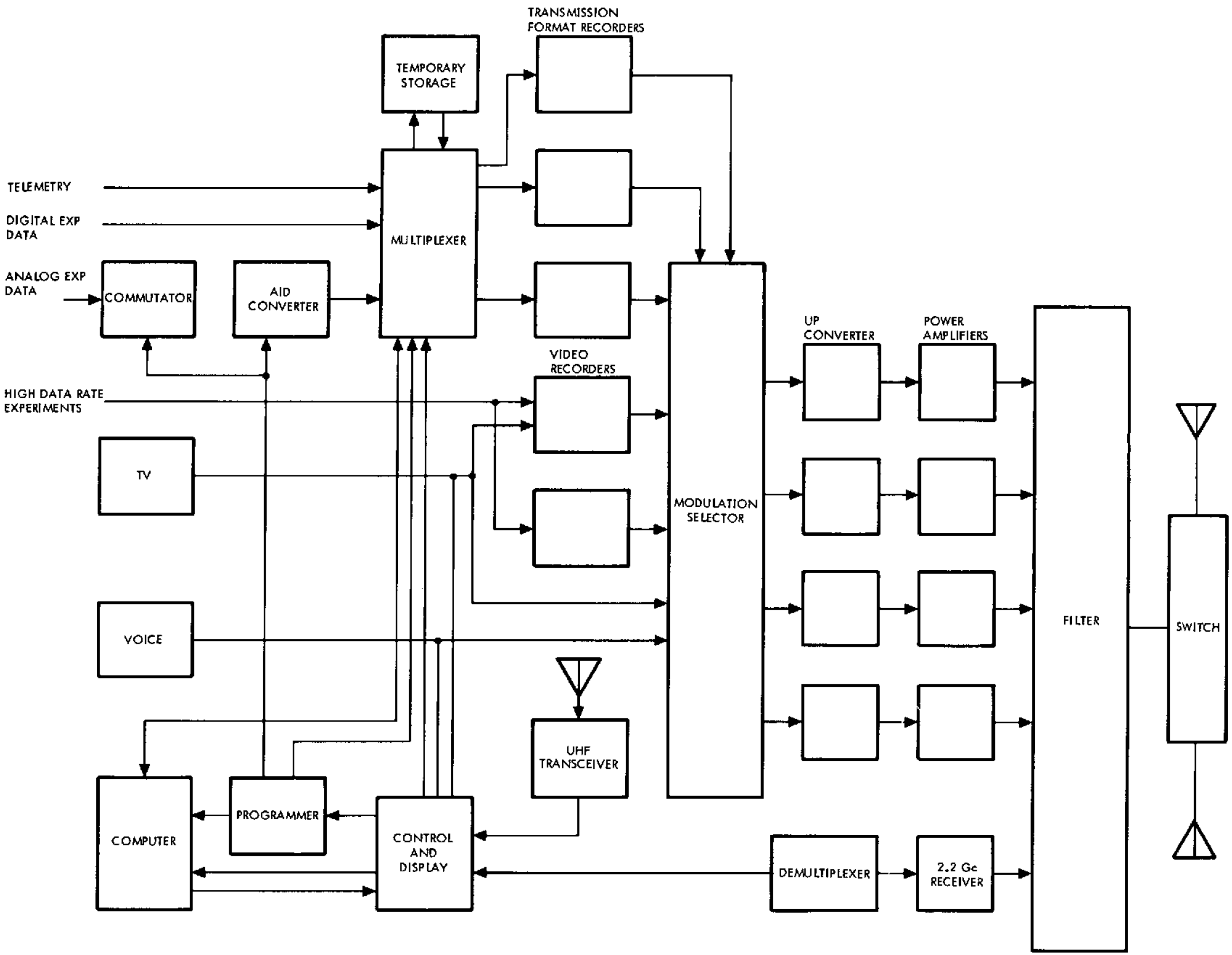


Figure 108. Block Diagram of the Space Station Communication System

Table 31. Estimated Weight and Power Requirements For Space Station Communication System

Equipment	Per Unit Wt. (lb)	Per Unit Pwr. (watts)	Units	Total Wt. (lb)	Total Power (watts) (Avg Use)
Crew Comm:					
1. Module Instr	8	4	6	48	24
2. Hub	8	4	1	8	4
3. Operations Center	2	-	-	2	-
4. Monitor	5	8	7	35	56
Television:					
1. Monitor	10	20	3	30	60
2. Cameras	5	8	6	30	48
3. Drives	10	20	1	10	20
On-Board Data Processing Recorders:					
1. Utility	10	20	4	40	20
2. Norm. Trans.	95	150	3	190	300
3. Video	85	150	2	170	150
Computer:					
1. Transmission Link	165	200	1	95	120
2. Power Amplifier	8	40	4	32	80
3. Modulators	5	5	4	20	10
4. Frequency translators	4	5	4	16	10
5. Frequency Sync	3	4	1	3	5
6. Diplexer	4	-	-	4	-
Receiver:					
1. 2.3 Gc	10	10	2	20	10
2. UHF/VHF	5	8	2	10	10
3. Antennas & Boom	5	-	2	-	-
4. Wiring, harnesses, support	-	-	-	170	-
Total				933	927

Both systems will have a 4-kc voice band width, will have access to the recorders, and will be able to be monitored from the operations centers.

Monitor System

This system, of course, is highly dependent upon the final laboratory system configuration; however, the system described is considered representative. Two classes of information are considered — simple monitor and maintenance information, and critical life-support information. Both types may be either in analog or digital format. The analog signals may be subcommutated by a low-level commutator before signal conditioning to reduce the signal-conditioning equipment required. The analog information will be fed to the primary commutator, then to an analog-to-digital converter. The resulting digital signal is then introduced to the main-time multiplexer, along with the data from the digital sensors and the sync code generated by the programmer. The output of the multiplexer goes to the central data-processing equipment and to the display facilities.

The life-support parameters are monitored in the same manner, but provision is made also for immediate display and danger warning whenever a given parameter falls outside pre-selected limits. Provision is also made to override the automatic programming with programs originating from the malfunction-investigation programs.

Each module will contain a system of this type. The exact configuration will vary from module to module, but in all cases the system will consist of the same components.

Television

The proposed television system is the one currently favored for Apollo, for which a great deal of effort is being expended to provide an adequate television presentation using reduced band widths. It is believed that adequate pictures may be obtained by using information band widths on the order of 2 megacycles. In this scheme the frame rate is reduced and the flicker problems are compensated by special interlace techniques. This mode of operation will be used throughout the space station. The on-board monitors thus will present the same information that can be transmitted to earth.

This system has been selected for several reasons. First, experiments have shown that transmissions of this type—having sufficient signal-to-noise ratios — provide entirely satisfactory displays. The picture is not an ultra-resolution presentation, but use of television for operations support, and as a public information service, does not require the ultimate in resolution. The second reason for using this particular form of television is the obvious desirability of using techniques already available. The use of these systems will reduce the development system test prior to use for lunar missions. Finally, though power for transmission is not a major problem for this application, the reduced band width facilitates on-board storage of the television and reduces the portion of the spectrum required for transmission.

The system has three video channels and three monitors in the space station; the three channels can be fed by any of six cameras. At present, two cameras with remote controls are planned for extra spacecraft observations, and four cameras are available for either permanent or semipermanent installations inside the lab. Various lens arrangements will be provided for the varying requirements, and further investigations may prove that sensors of different sensitivity are required. In this case, the cameras will be designed so that replacement is possible.

The slow-scan television is obtained from special cameras having extremely high resolution capabilities. These cameras will have several exposure times available to provide the correct exposure. Once exposed, the pictures will be scanned at some nominal rate consistent with the other processes in the space station. The resulting signal may be transmitted directly to earth, but most probably will be recorded for subsequent transmission. (The selection of format remains to be determined when the recording and transmission formats become more firm.)

On-Board Data Processing

This system performs all control, storage, and programming services; and, hence, the system forms the operations center for the station. This discussion of the system is divided into three main sections: (1) the recording devices, (2) the programmer, and (3) the general-purpose computer.

Recording Devices

Three types of recorders are under consideration for use on the station. One would serve as a general-purpose unit, accepting analog or digital data from the various sources and storing it until it has been edited or evaluated by the operators. The second type of recorder would be used to assemble the data gathered into the format ready for transmission to earth. Finally, the third type of recorder will be capable of very wide band width recording (e.g., what is usually referred to as a video recorder).

Although the final operational requirements will determine the actual complement, for the present it will be assumed that four general-purpose recorders will be included. These units are to be used for general support of experimentation and operations, and will be rack-mounted but capable of being installed at whatever location they are required. They will have a 20-kilocycle band width on each of five information channels, and will use 2400-foot reels of 1/2-inch tapes to record at speeds of 1-7/8, 3-3/4, and 7-1/2 inches per second (the full band width need be realized only for the fast-record speed). For transmission to earth, information that has been recorded on these machines will be sampled and (if analog) put into a digital format for incorporation into the transmission book.

The recorders required for formulation of the transmission book present a somewhat more difficult problem. The target of the system design will be to deliver the estimated 2×10^9 bits of digital information collected during an average orbit to the transmission system during a 5-minute period. This appears to be a demanding but not prohibitive rate. Present storage density, numbers of parallel channels, and tape speeds can deliver this rate from a single-tape deck operating with stationary heads while maintaining reasonable bit-drop-out rates.

The final configuration selection must again await further data information, but the following parameters would typify such a machine:

Packing density	-	2×10^3 bits per inch
Tape speed for readout	-	120 inches per second
Number of channels	-	32

These recorders will have only two speeds, for simplicity. The loading will be done at a low speed, perhaps 1/32 or 1/64 of the read speed. All editing and data processing will be performed before the data are recorded. (The compilation of the transmission book will be the sole function of these recorders.)

The wide-band recorder presents an even greater problem. At present, there are magnetic tape recorders capable of 4-mc recording band width. These machines are not very flexible, require large amounts of power, are heavy, and are so intricate that their reliability is questionable for space applications. For this wide-band recording, therefore, a relatively new technique—thermosplastic recording—is suggested units have been developed that operate at 10-mc band widths, and a 50-mc capability is envisioned by 1965. It appears that a recorder having reliability characteristics at least equal to present techniques can be provided during the available time; if so, a great saving in power and weight, and a great increase in storage capacity and band width capability, can be realized. It is felt that incorporation of such a recorder is the most realistic means of acquiring a significant amount of wide-band recording, and subsequent transmission by the 1966 time period.

General Purpose Computer

The requirements given in the former sections are entirely too general to allow a definitive solution of the computer problem. There are several general-purpose computers in existence that are essentially applicable to this mission, but a special design will be required to provide the particular logic and packaging required. The special logic and the arithmetic section will be designed to accommodate the particular programming and data-reduction functions that will be necessary. These same requirements will dictate the working and program memories.

One of the primary uses of the computer will be the programming of the various data sources into the transmission book. This will include the scheduling of operations, the direction of data into temporary storage, and the collection of the various data sources into a format that makes optimum use of the on-board storage facilities.

Transmission Links

This section describes the r-f transmission links that appear to be most practical for the space station. It will be seen that the system provides a great deal of flexibility and reliability through multiple use of identical units.

The transmission equipment consists of four separate channels, each composed of a modulator, frequency translator, and power amplifier. The units are identical in all cases, with separate channels differing only in the base frequency fed to modulators, and in the form of the signal modulation.

Under normal data loads, one of the four transmitters will carry most of the data. Frequency multiplexed into this carrier will be 2×10^9 bits of stored digital data, four voice links, and the various timing of the sync signals that are required. This information can be multiplexed into an 8-mc information band width. The resulting r-f signal can be transmitted with adequate received signal-to-noise ratios when a 20-watt transmitter is used in conjunction with an omnidirectional antenna on the space station and a 60-foot parabolic antenna at the ground station.

A second transmitter, also modulated with an 8-mc information band width, is normally used to transmit two channels of television information.

These two transmitters will handle the normal data loads; however, the remaining two transmitters will be used to handle the high-data-rate-sensor information, data acquired during orbits during which the ground station is inaccessible, or in the event of a transmitter failure. These channels would again nominally accommodate an 8-mc information band width, so the transmitter center frequencies could be located at 2208 mc, 2226 mc, 2244 mc, and 2262 mc.

As previously noted, any of the transmitters may be used to handle any of the data formats; however, the lowest frequency will normally handle the routine data, the next transmitter the television, and the last two transmitters will handle the wide-band stored data.

Antenna System

Several antenna systems were investigated for the space station. The primary difference involves the choice between high-gain or quasi-isotropic antennas. However, if the 60-foot antennas of the TLM-18 system are assumed for the ground station, the above transmissions can be accomplished with more than ample safety margins without employing directional antennas on the spacecraft.

It is therefore recommended that two quasi-omnidirectional antennas be employed: one deployed from a module parallel to the station spin axis and toward the sun; the other deployed in the opposite direction. Tests would be necessary to ascertain the effect of the station in the near field of the antenna, but this antenna configuration would seem desirable since use of high-gain tracking antennas would increase system complexity and become a source of reliability degradation.

Ground Station

The two ground stations proposed to support the space station will each provide complete capability for up-link transmission, local display, recording, data reduction, and a real-time data link with the command center in Houston. The following describes in detail some of the more pertinent features of the station diagrammed in Figure 109.

The ground station will use a modified TLM-18 antenna employing a wide-band feed for the 2.2- to 2.3-g band. A duplexer isolates the transmitter and receiver signals, the latter being amplified by a wide-band TWT amplifier. The output of this amplifier is subsequently demultiplexed into the four receiver channels and detected by an individual receiver. The outputs of the receivers are then routed through the control console for subsequent recording and display or retransmission.

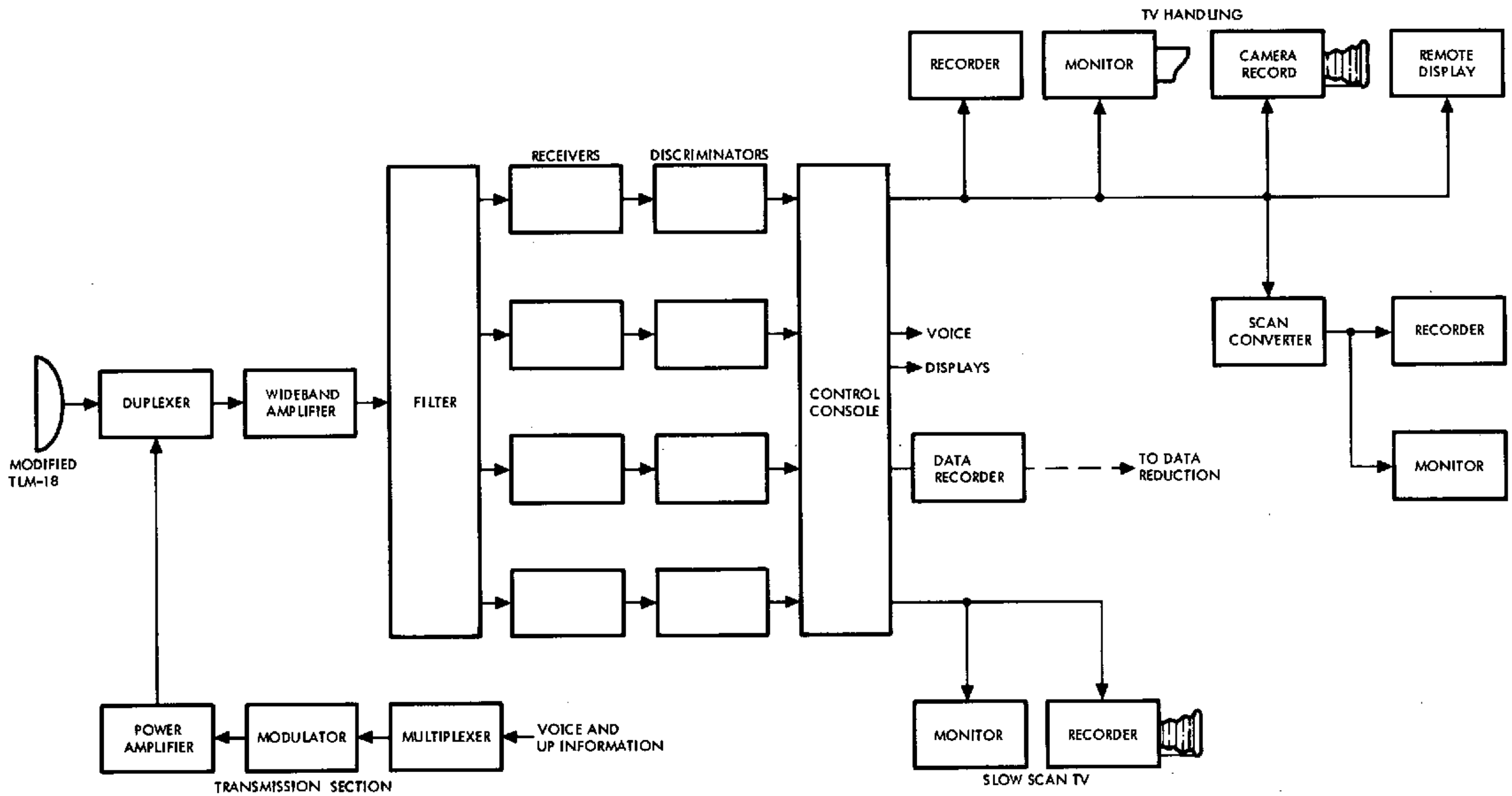


Figure 109. Block Diagram of the Ground Station Configuration

The television channels are routed to the recorders, monitors, and scan converters. Both recording and quick-look display are provided for the raw television picture. In addition, the signal is changed to standard commercial television format by the scan converter unit. This picture is again monitored and recorded. The pictures are subsequently edited and distributed to the using agencies.

CONCLUSIONS

There seems to be little question as to the feasibility of providing a very capable system within the assumed development time. The only real question appears to center on the required wide-band recorders; and even here, development is required only to lighten the system penalties imposed by the requirements. The following paragraphs, therefore, outline studies that will be undertaken to further refine the system configuration (rather than to establish basic feasibility).

Ground Station Configuration

The final ground station configuration may be quite different than the rather restrictive one assumed in this report. Further study is recommended to investigate the space station operations concept (it is quite possible that the long time between communication periods may compromise some presently unrecognized requirements), evaluate future ground-station plans for Apollo and other space programs, and to refine further the specifications for the ground-station equipment.

Wide-Band Recorder

While there seems to be little doubt that the selected technique will have reached a sufficient degree of sophistication during the development period, a more detailed examination will be made of specific uses for this recorder and of alternate possibilities which might provide backup capability.

Spectrum Utilization

One of the greatest differences between this and other space programs is the quantity of data to be collected from the space station. The use of only a few primary ground stations requires that the space-to-earth transmission link be sufficiently wide-band. There are, in fact, no specific allocations for space use to accommodate such transmissions. It is presently assumed that the 22-to 23-gc telemetry band will be used, but it is desirable to minimize spectrum usage as much as possible (consistent with fulfilling the needs of the space station) to permit allocations for other programs. Therefore, the various data format and modulation techniques will be analyzed more closely to determine their particular applicability to this problem. Consideration will also be given to higher frequency bands where spectrum occupancy will not be as pressing a problem.

On-Board Data Processing

This is a very complicated facet of the communication system problem for the space station and will require a great deal of investigation. Problems involving programming, data reduction, compacting, and the transmission-book format all require detailed study, being highly dependent upon operation concepts and data qualities as yet unestablished. Substantial effort is planned to eliminate these problems by the establishment of more definitive system requirements.

Antenna Systems

Some problem is anticipated in providing the quasi-omnidirectional coverage proposed in this report. A more thorough investigation of the effects of the spacecraft structure is planned now that the station's configuration has been firmly established.

WEIGHT ANALYSIS

During the six-month feasibility study, space station weights were defined repeatedly in order to provide preliminary weights, center of gravity, and moment-of-inertia data for use in structural and stabilization analyses. The results of the final calculations — using the recommended systems — are presented in this section. Structural group weights, equipment group weights, and a weight summary are shown in Tables 32, 33, and 34. Table 35 contains center-of-gravity and inertial data.

The weight of the total Saturn C-5 payload (space station plus one Apollo) at launch is 170,300 pounds. The space station structure, with a weight of 75,150 pounds, accounts for approximately 50 percent of the total launch weight. Other equipment in the space station weighs 63,450 pounds, giving a total space station weight of 138,600 pounds. The addition of the Apollo spacecraft including command module, service module, three-man crew (15,900 pounds), and the launch escape system (5800 pounds — and interstage structure (10,000 pounds) between the S-II booster stage and the space station yields the total launch weight of 170,300 pounds.

Table 32. Structural Group Weight Summary

Structural Groups	Weight (lb)	
Hub		15,590
Apollo Stowing Turret:	(5057)	
1. Basic structure	3123	
2. Booms and operating mechanism	684	
3. Spindles	300	
4. Adapters (6)	750	
5. Turret rotating mechanism	100	
6. Equipment supports	100	
Central Compartment:	(10533)	
1. Basic structure	3670	
2. Zero gravity laboratory	200	
3. Passage to Spokes	1473	
4. Passage to Apollo spacecraft (incl. 7th adapter)	895	
5. Boost flare	3204	
6. Support structure	891	
7. Equipment supports	200	
Spokes (3)		8,860
Honeycomb Shell	6600	
Longerons	580	
Frames	120	
Expandable joint and seal	600	
Pressure lock and module junction (3)	420	
Hub junction and seal (3)	150	
Ladders	60	
Deployment mechanism and locks	270	
Equipment supports	60	

Table 32. Structural Group Weight Summary (Cont.)

Structural Groups		Weight (lb)
Modules (6)		50,700
Honeycomb shell	30800	
Longerons	5700	
Frames	600	
Floor	4800	
Spoke junction (3)	300	
Module junction (6)	3600	
Equipment supports	4000	
Deployment mechanism and locks	600	
Launch attach fittings	300	
Total		75,150

Note: Interstage structure of 10,000 lb. is shown in Table 34 to arrive at a total payload structural weight of 85,150 lb.

Table 33. Equipment Group Weight Summary

Equipment Group	Weight (lb)			
	Hub	Spokes	Modules	Total
Instruments and panels	24		144	168
Attitude Control:			(3070)	(3070)
1. Propellant			2350	2350
2. Tanks and plumbing			520	520
3. Jets and electronics			200	200
Wobble Control			330	330
Electrical Power System:	(1341)		(9939)	(11280)
1. Power for Oxygen Regeneration	730		4380	5110
2. Power for Remaining Systems	(611)		(5559)	(6170)
3. Batteries:	258		2314	2572
a. Chargers	25		200	225
b. Solar cell panel	248		2310	2558
c. Inverters	30		270	300
d. Regulators			15	15
e. Wiring, lights, misc.	50		450	500
Communications	34		1060	1094
Laboratory equipment (incl. wiring, plugs, misc.)			21000	21000
Environmental Control System:	(2386)	(1054)	(19536)	(22976)
1. CO_2 removal units	64		384	448
2. Air glycol heat exchanger	25		150	175
3. Charcoal bed	15		90	105
4. Blower and spare	26		156	182
5. Odor-control unit	10		100	110
6. Water receiver and separator	5		50	55

Table 33. Equipment Group Weight Summary (Cont.)

Equipment Group	Weight (lb)			
	Hub	Spokes	Modules	Total
7. Glycol, pump and spare	6		36	42
8. Space radiator	100		600	700
9. O_2 Glycol heat exchanger	10		60	70
10. Valves	21		126	147
11. Misc.	6		72	78
12. Nitrogen Tanks	300	410	3312	4022
13. Nitrogen	150	195	1584	1929
14. Oxygen Tanks	193	103	2100	2396
15. Oxygen	77	257	840	1174
16. Liquid Oxygen Tanks	244		1464	1708
17. Liquid Oxygen	504		3024	3528
18. Instrumentation	24		144	168
19. Supts.; plumbing, wiring, misc.	180	89	1428	1697
20. Oxygen regeneration system	(256)		(1536)	(1792)
a. CO_2 reactor	40		240	280
b. Electrolysis cell	216		1296	1512
21. Air sealed in modules and hub at launch	170		2280	2450
Personnel Accommodations:			(3532)	(3532)
1. Toilets (3)			168	168
2. Wash basins and showers			84	84
3. Food preparation			132	132
4. Dispensary			21	21
5. Washing Machine			66	66
6. Clothing			63	63
7. Bedding and bunks			150	150
8. Food			1362	1362
9. Food containers and storage			166	166
10. Partitions, tables and chairs			219	219
11. Water			656	656
12. Tanks			25	25
13. Misc. personal items			210	210
14. Utility and waste water system			210	210
Total	3785	1054	58,611	63,450

Table 34. Space Station Weight Summary

Space Station	Weight (lb)			
	Hub	Spokes	Modules	Total
Structure	15,590	8860	50,700	75,150
Equipment:	(3785)	(1054)	(58,611)	(63,450)
1. Instrument and panels	24		144	168
2. Attitude controls			3070	3070
3. Wobble control			330	330
4. Electrical power system:	(1341)		(9939)	(11,280)
a. For oxygen regeneration	730		4380	5110
b. For remaining systems	611		5559	6170
5. Communication	34		1060	1094
6. Laboratory equipment			21,000	21,000
7. Environmental control system	2386	1054	19,536	22,976
8. Personnel accommodations			3532	3532
Total at Launch	19,375	9914	109,311	138,600
Plus:				(31,700)
Launch escape system				5800
Apollo and 3-man crew				15,900
Interstage structure				10,000
Total Boost System Payload				170,300

Table 34. Space Station Weight Summary (Cont.)

Space Station	Weight (lb)			
	Hub	Spokes	Modules	Total
Less:				(-31,700)
Launch escape				-5800
Interstage structure				-10,000
Apollo and 3-man crew				-15,900
Plus:				
Apollos (7 and 21 crewmen)	106,680		4620	+111,300
Total Space Station Laboratory Deployed, Fully Manned and 7 Apollos Stowed	126,055	9914	113,912	249,900

Table 35. Preliminary Weight, Center-of-Gravity, and Inertia Data

ITEM	Weight (lb)	CG (1) x	CG (1) y	Ix Slug Ft^2	Iy Slug Ft^2
Base Platform	138,600	-10.6	0	15,250,000	7,750,000
Base Platform and 1 Vertical Apollo (3-man crew) in 1 Module	154,500	23.6	2.5	15,321,000	7,760,000
Base Platform and 1 Horizontal Apollo (3-man crew) in 1 Module	154,500	5.24	22.3	15,440,000	7,800,000
Base Platform and 7 Apollos (21 crewmen) Distributed in all modules	249,900	69.2	0	16,700,000	9,100,000
Hub-Apollo Stowing Turret - No Apollo	6,200	194.0	0	5,265	5,239
Hub-Apollo Stowing Turret + 1 vertical Apollo	21,500	294.3	0	15,865	37,300
Hub-Apollo Stowing Turret + 1 horizontal Apollo	21,500	162.7	142.3	56,427	58,250
Hub-Apollo Stowing Turret + 2 horizontal-Opposed Apollos	36,800	157.4	0	295,500	279,600
Hub-Apollo Stowing Turret + 7 Apollos	113,300	177.4	0	688,000	888,100

Table 35. Preliminary Weight, Center-of-Gravity, and Inertia Data (Cont.)

ITEM	Weight (lb)	CG (1) x	CG (1) y	Ix Slug Ft^2	Iy Slug Ft^2
Hub, complete - No Apollos	19,375	111.6	0	28,000	44,000
Hub, complete and 7 Apollos (no crew)	126,000	167.0	0	910,000	635,000
Total Weight Earth Launch Booster Payload	170,300	(2) 691.6	0	590,000	7,500,000
NOTE: (1) ℄ of stowed Apollo is x = 150 vertical ℄ of station is y = o (2) From booster, payload interface					

HUMAN FACTORS

This section describes the self-deploying space station design features which would be imposed by human factors requirements. The nature of the artificial gravity environment that is acceptable for human habitation is the major item of discussion. Other factors will be described, such as the relationship between the visual and gravity environments, safety considerations, and general design features and environmental envelopes that will determine the extent of a tour of duty. In addition, recommendations are discussed for research programs required to resolve the tentative solutions indicated in this report.

DESIGN PROBLEM

The purpose of a human factors investigation of the self-deploying space station is to establish a design envelope that is satisfactory with respect to the effect of the induced gravity environment on the crew. The problem is the creation of a design which will permit the crew to work with comfort, efficiency, and safety. In that providing a favorable environment for the human occupants of a space station is the primary justification for creating artificial gravity,it is necessary to ensure that the induced environment does not create more problems that it solves. The psychological, medical, and physiological literature shows many examples of severe discomfort, major decreases in the performance of operators, and sickness — all produced by the experience of unusual gravity environments. However, it appears that there is a design envelope within which the gravity stimuli may be controlled to avoid the undesirable responses.

Artificial Gravity Environment

The artificial gravity is the centrifugal force in a rotating, nonaccelerating space station. A man's weight can therefore be expressed as the centrifugal force vector and has a magnitude equal to $(\omega^2 r)\ /g_c$ where omega (ω) is the angular velocity of rotation expressed in radians per second and r is the perpendicular distance from the axis of rotation to the site of the action of the force. The gravitational constant g_c (32.2 ft/sec^2) is used to express the force in terms equivalent to normal experiences on the earth's surface. In the case of the rotating station, the force acts at the man's feet and is expressed as the distance from the axis to the most distant inner surface upon which a man may be standing or sitting. It is evident that if the station occupants could remain at a constant position, small values of ω

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could be used in the station design. However, the advantages of the artificial gravity would be lost if the crew were required to be immobilized.

Gravity Gradient

The layman would generally think of an artificial gravity environment as being a fixed value for all objects in a space station. However, closer examination of the situation shows that there is a gravity gradient perpendicular to r. If the man is standing, his head would weigh less than his feet. If he moved up or down, as in rising from a reclining position, his head would vary in weight. It is clear, that at small radii, extreme variations in the force acting on the man's gravity receptors would occur as he moved about. The lower acceptable radius can be partially defined relative to a tolerable gravity gradient.

The percentage of head-to-foot variation can be expressed as the ratio of the man's height to the radius which gives the proportion of variation which would occur. Figure 110 shows percent of head-to-foot variation for a 6-foot man as a function of station radius. Inspection of Figure 110 shows that at a 40-foot radius, the rate of change of head-to-foot difference changes and smaller advantages are gained by equal increases of radius. This factor allows an engineering compromise to the minimum radius requirements. The occurance of the "break" at a 15-percent g gradient has led some writers to suggest that 15 percent is the minimum with respect to human tolerances. However, the sole use of the gravity gradient as the source of the criterion for a minimum radius is not acceptable due to the following factors:

1. It is relevant only to a stationary man, in that it combines with other considerations when motion occurs.

2. The gradient can be reduced by body orientation, i.e., at right angles to the axis of rotation.

Gravity Variation

Sole use of a potential gravity gradient on the immobile crew member was shown to be unacceptable for the definition of the minimum radius. A major value of the rotating station is that it enables the crew to move in a relatively normal manner. An analysis of the factors which directly influence man's ability to move comfortably shows that coriolis forces and rim speed are the factors of concern. In addition there is a combined effect of coriolis forces and the level of artificial gravity.

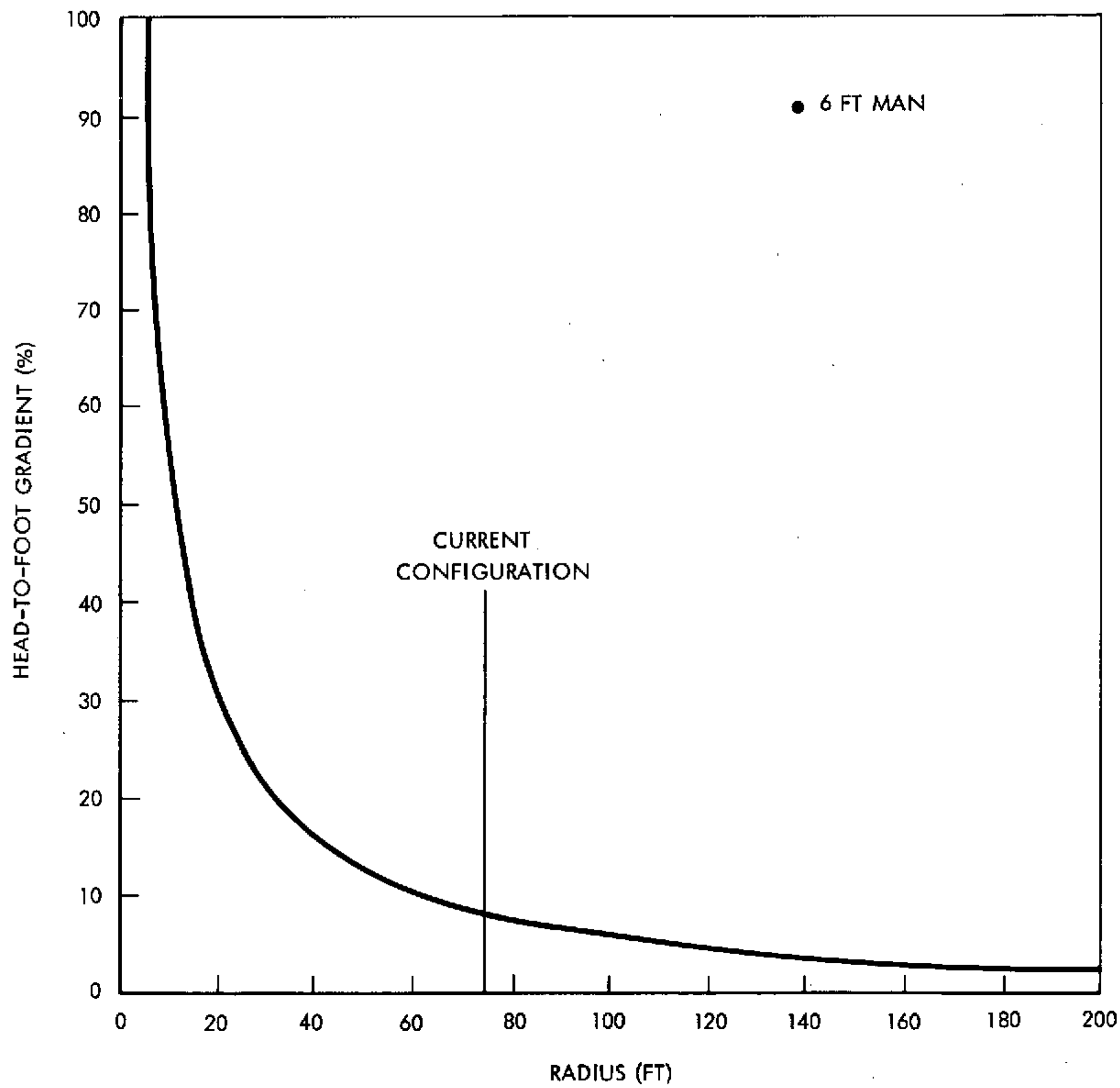


Figure 110. Gravity Gradient Versus Radius

The ratio of coriolis force to centrifugal force is

$$\frac{\text{Coriolis force}}{\text{centrifugal force}} = \gamma = \frac{2M(\omega_V F_m)}{M\omega_V(\omega_V R_V)}$$

where

M = mass of man

ω_V = angular velocity of vehicle

R_V = radius of rotation

V_m = velocity of man relative to vehicle

Assuming man's velocity to be perpendicular to the rotation plane and the rotation radius perpendicular to the rotation axis, then

$$\gamma = \frac{2V_m}{\omega_V R_V}$$

Such a relationship demonstrates that the ratio varies directly with the velocity of man (walking) and inversely with the rotation radius and rotation rate. Increasing the ratio may cause two effects on man: an increase or decrease in man's total weight, and/or radical change in man's subjective sense of down and by the generation of visual illusions. Illusions may occur with movement in a radial plane, such as rising from a prone or sitting position, and the total weight change when movement is along the floor. The second factor, rim velocity (V), of a rotating vehicle is

$$V = \omega R$$

where

ω = angular velocity

R = radius of rotation

The centrifugal acceleration, A, due to uniform circular motion is

$$A = \frac{V^2}{R}$$

where

V = tangential velocity

R = radius of rotation

The rim velocity is important since it is involved in the effects on man of walking with or against the direction of rotation. Variations in g loading may impose many problems if permitted to approach a significant percentage of normal gravity. The acceleration ratio of man moving, g_m, to a man stationary with the vehicle rim, g_s, is

$$\frac{g_m}{g_s} = \left(\frac{V_m}{V_s}\right)^2 = \left(\frac{V_s \pm v}{V_s}\right)^2$$

Assuming a uniform radius:

V_m = absolute velocity of moving man

v = speed of movement relative to vehicle

V_s = absolute velocity of stationary man (on rim)

From this we see that if a man walks fast enough to equal the spin of the vehicle, depending on the direction, he may either reduce his g loading to zero or increase it 400 percent. It is believed that an extreme limit for the radial acceleration experienced by a man moving against rotation to a man at rest may be set at 0.5 or 50 percent. This relationship is given as

$$\frac{(V_s - v)^2}{V_s} \geq 0.5$$

where v is taken as unity, the quadratic solution yields

$$V \geq 3.4(v)$$

where V = rim velocity

Assuming a walking velocity of 3 feet per second, a rim velocity of 10+ feet per second is required. This does not, however, take into consideration the ratio of the Coriolis force to the centrifugal force. Assuming a centrifugal to coriolis ratio γ of 0.25, then

$$\left(\frac{V_s - v}{V_s}\right)^2 - 0.25 \geq 0.50$$

$$\left(\frac{V_s - v}{V_s}\right) \geq 0.75$$

$$V_s + V \geq 7.7_v$$

If a walking speed of 3 feet per second is assumed, $\underline{V} \geq 23.1$ feet per second. Normal speeds vary from under 2 or 3 feet per second while walking to as much as 30 feet per second while running. Under condition of reduced g, it is questionable that man can or would want to run at high speeds. Hence, the recommended rim speed (3.4 v) is at least 10 feet per second and preferably nearer 20 feet per second. This velocity of 3 feet per second would also produce radial acceleration changes of approximately 0.05 g in a 150-foot-diameter station rotating at 3 rpm. Based on extrapolations made from the slow rotation room studies, 8:1 is considered tolerable.

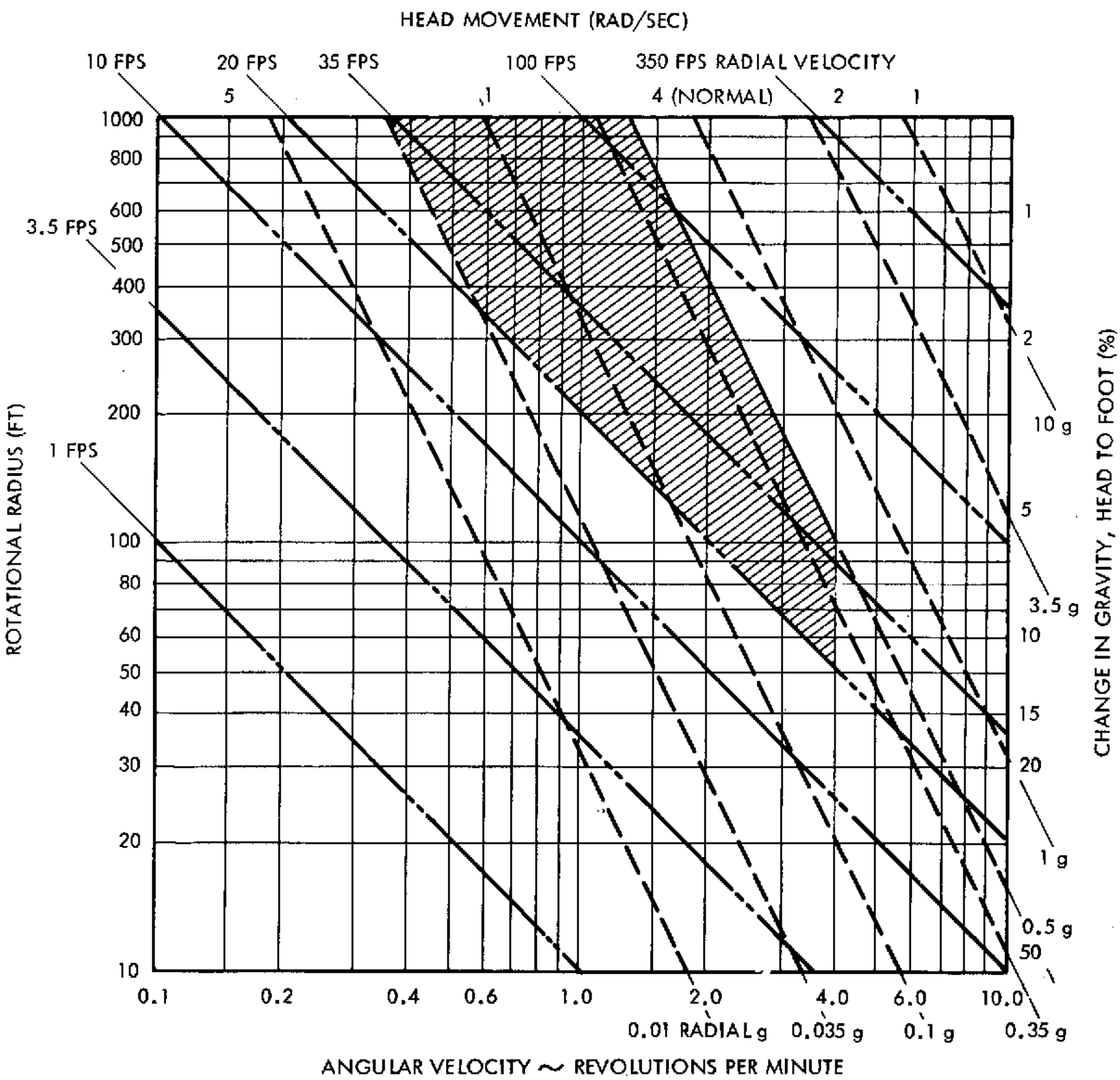
Gravity Limits

The physical relationship that exists between the velocity vectors during rotation have been described by Dole (Reference 1) and are shown in Figure 111. The upper limit of gravity is reasonably specified at 1 g. Experimental evidence with values above 1 show that excessive g results in muscular fatigue and degraded performance. Further, all aspects of $g \geq 1$ have by no means been investigated. The useful lower limit of g is not known. However, there is good reason to believe that some practical lower g limit exists below which the g value would be too low to be useful. Estimates of the lower g limit range as low as 0.01 g. However, in view of the conspicuous lack of experimental data that presently exists, it seems advisable to use a higher g value, i.e., 0.20 g, as a more suitable lower limit in defining a usable g region. This value is the tentative lower limit at which walking can be done without aids, such as adhesive shoes. In the discussion of rim velocities in the previous section, recommendations of rim velocities between 10 feet per second and 20 feet per second were made. A rim velocity of 20 feet per second, imposes a further restriction on the lower g limit as shown in Figure 111. Finally, the design envelope recommended is for values as high as can be achieved without producing motion sickness at reasonable head rotation rates ($\theta = 4$ is the normal rotation rate).

Coriolis Acceleration

The sensory organs which determine head position and sense motion consist primarily of the vestibular apparatus and the eyes. The vestibular apparatus responds to forces of acceleration while the eyes sense relative position and motion of the head with respect to visual surroundings. The geometric relationship of a man on a rotating platform is shown in Figure 112. Taking a rotary platform of radius N, axis of rotation Z, rotational rate of ω radians per second (constant), and a man seated at A on the platform edge facing the axis of platform rotation, the rotational rate about point A is θ (constant), and the rate of change of r is given by the rate at which AD changes (Figure 113):

$$AD = h \cos \lambda$$



NOTE: ADAPTED FROM DOLE REFERENCE 1.

Figure 111. Rotation Effects

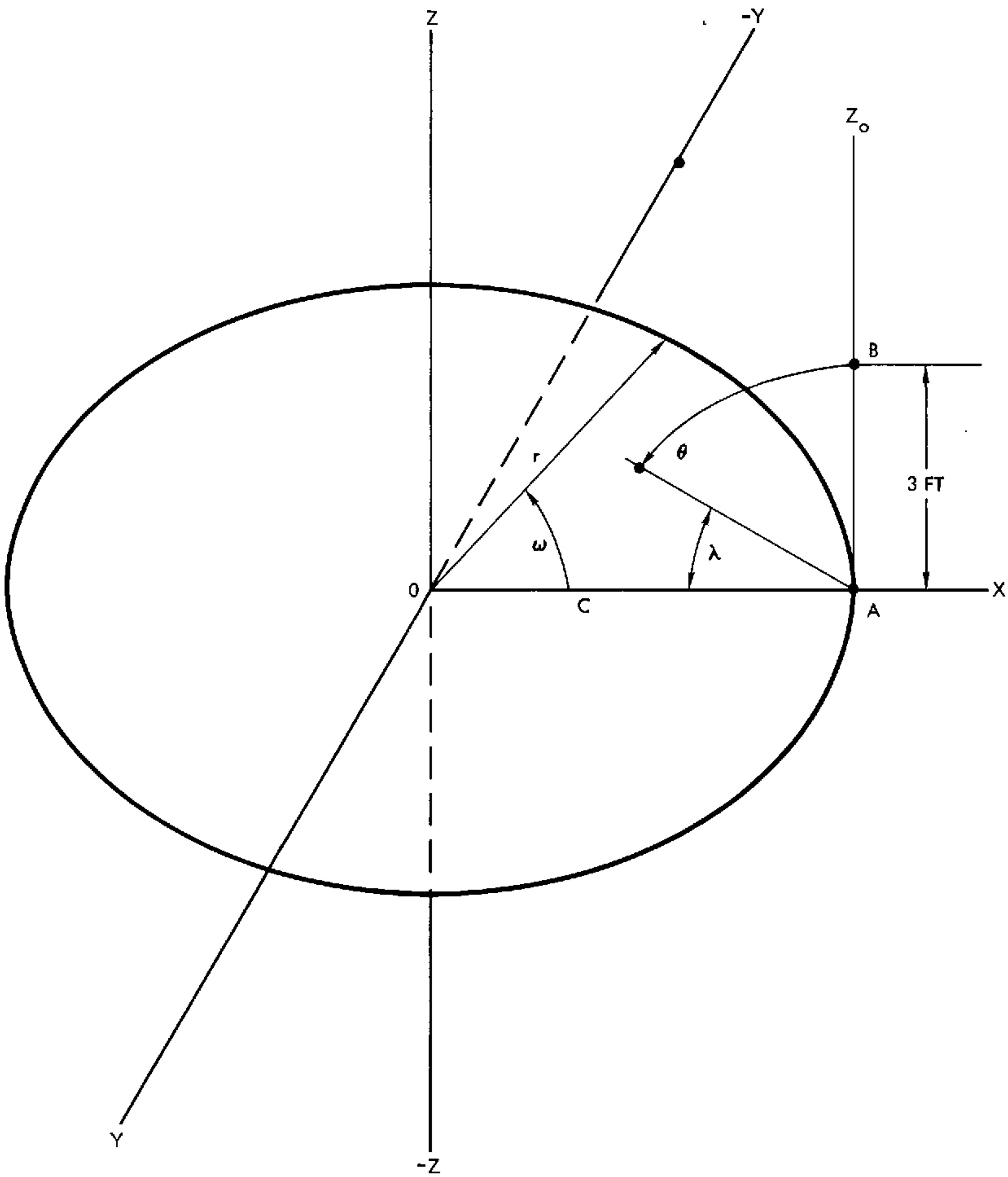


Figure 112. Geometric Relationship - Rotary Platform and Moving Person

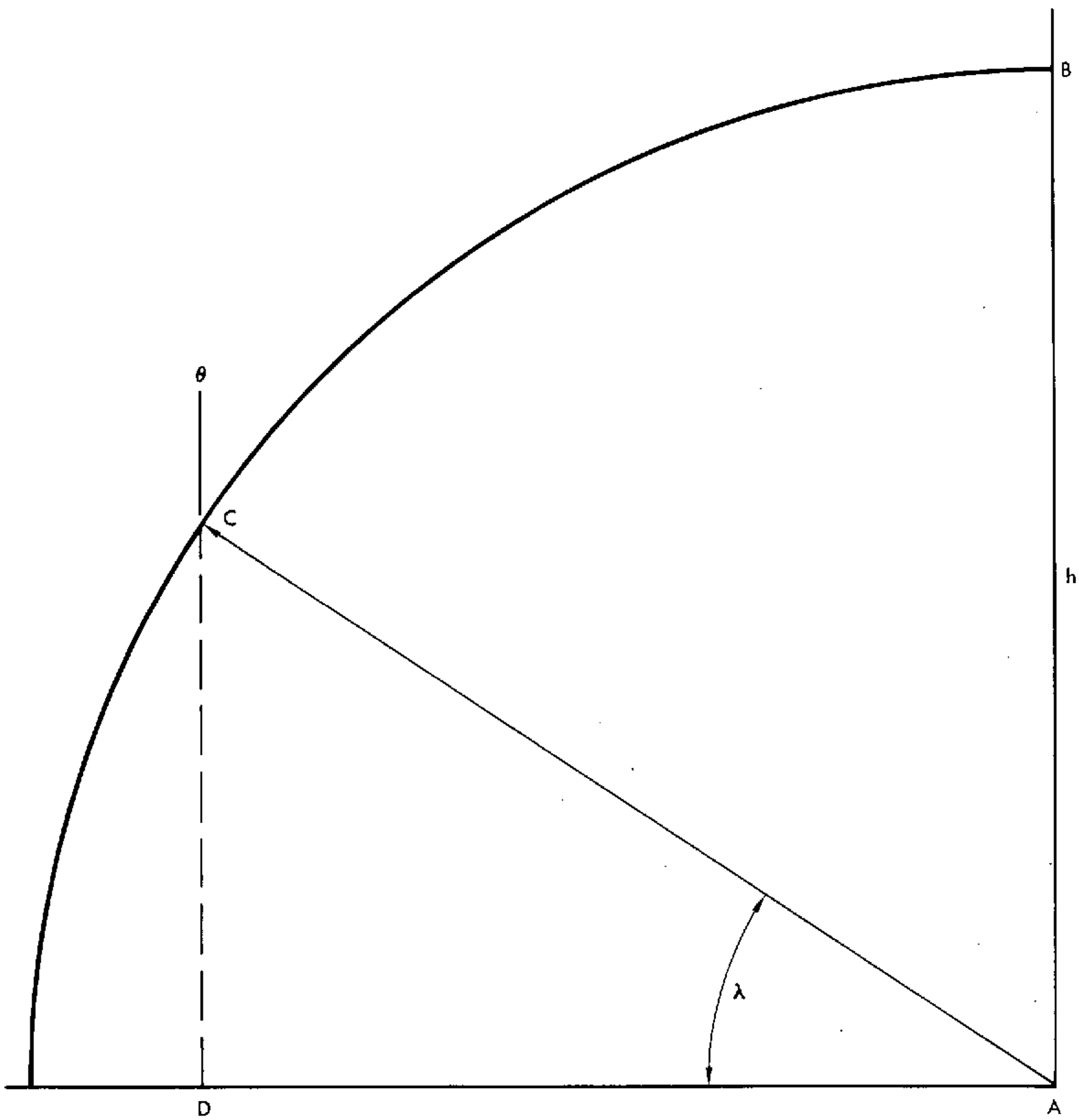


Figure 113. Plane of Body Rotation

$$\frac{dr}{dt} = h \sin \frac{\lambda d}{dt}, \text{ but } \frac{d\lambda}{dt} = 0$$

therefore

$$\frac{dr}{dt} - h0 \sin \lambda$$

$\frac{dr}{dt}$ is at maximum when the angle λ = 90 degrees minimum when λ = 0 degrees

Since the radial velocity of point c about the Z axis is ωr then the rate of change of velocity V_c with time of a man's head due to rotary motion θ is a type of Coriolis acceleration A_{cor}, that is

$$\frac{dV_c}{dt} = \frac{\omega dr}{dt} = h\omega\theta \sin \lambda$$

This equation represents an acceleration which is also maximum when λ = 90 degrees. In man, the h for a sitting position is: $h \simeq 3$ feet. Therefore: $A_{cor} = -3\omega\theta \sin \lambda$.

There is evidence which indicates that the endolymph displacement in the semicircular canals is proportional to the angular velocity of the head; therefore we may assume the angular acceleration $\frac{C\theta}{dt} = 0$ and not consider this effect.

The negative sign of the Coriolis acceleration yields the direction of the vector for Coriolis deflection. Under conditions of counter-clockwise rotation ($+\omega$), body rotation from the periphery toward the center ($+\theta$), produces a deflection to the right (parallel to and in the direction of -Y). Reversing either the direction of ω or θ reverses the direction of the Coriolis deflection. Attention must be directed toward the fact that most data on Coriolis effects for values of ω other than 1 is extrapolated from experimental data where the angular rate is $\omega = 1$. Another Coriolis force is experienced when an object is caused to move radially toward or away from the axis of rotation at a uniform linear velocity. This acceleration is

$$a = 2\omega v \sin \phi$$

where

V is the linear velocity of the object

ϕ is the angle formed by the velocity vector and radius.

For maximum effect, ϕ is taken at 90 degrees. With normal arm movements of from 5 to 15 feet per second, accelerations are larger than those experienced in trunk movement (Figure 114).

However, man is used to working against g forces of even larger magnitudes in normal earth gravity; therefore it is not anticipated that such forces will constitute a problem. The limiting effects of the acceleration of Coriolis force on the vestibular apparatus is the decisive factor in vehicle rotation considerations.

A summary of artificial gravity design criteria has been completed by Loret (Reference 2). Figure 115 is adapted from this work and can be seen to be in close agreement with Dole's recommendations as was shown in Figure 111. In general, specialists concerned with the provision of a comfortable g environment have relied on a series of experiments at the U.S. Naval School of Aviation Medicine at Pensacola, Florida. Reference to the work of A. Graybiel, as found in the bibliographies of References 1 and 2, should be made for the details of these experiments. In the past years, experiments have been conducted on the human centrifuge at the U.S. Naval School of Aviation Medicine, Pensacola, Florida, in an effort to determine tolerance of human subjects to prolonged confinement in a rotating environment similar to that which might be encountered in a rotating space station. Groups of subjects lived for two days in a multisided (nearly circular), windowless room constructed around the center post of the human centrifuge. The room is 15 feet in diameter and 7 feet high and may be driven at constant angular velocities at rates from 1.7 to 10 rpm. Angular velocities used for the test were 1.71, 2.21, 3.83, 5.44, and 10 rpm. Certain general conclusions were as follows:

1. The results of the study gave supportive evidence to the notion that prolonged constant rotation and resulting head movements did not have a lasting deleterious effect on task performance although subjects were exposed to substantial amounts of stress particularly at higher velocities.

2. The dominant stimulus in the situation is the bizarre intermittent stimulation of the semicircular canal which results from motions of the head.

3. Adaptation was observed with all subjects in all runs of the test. Unpleasant symptoms associated with canal sickness, which were experienced earlier in the run, tended to disappear with time so that performance and subjective reactions noticeably improved for some subjects by the end of the first run and for all subjects by the end of the second run.

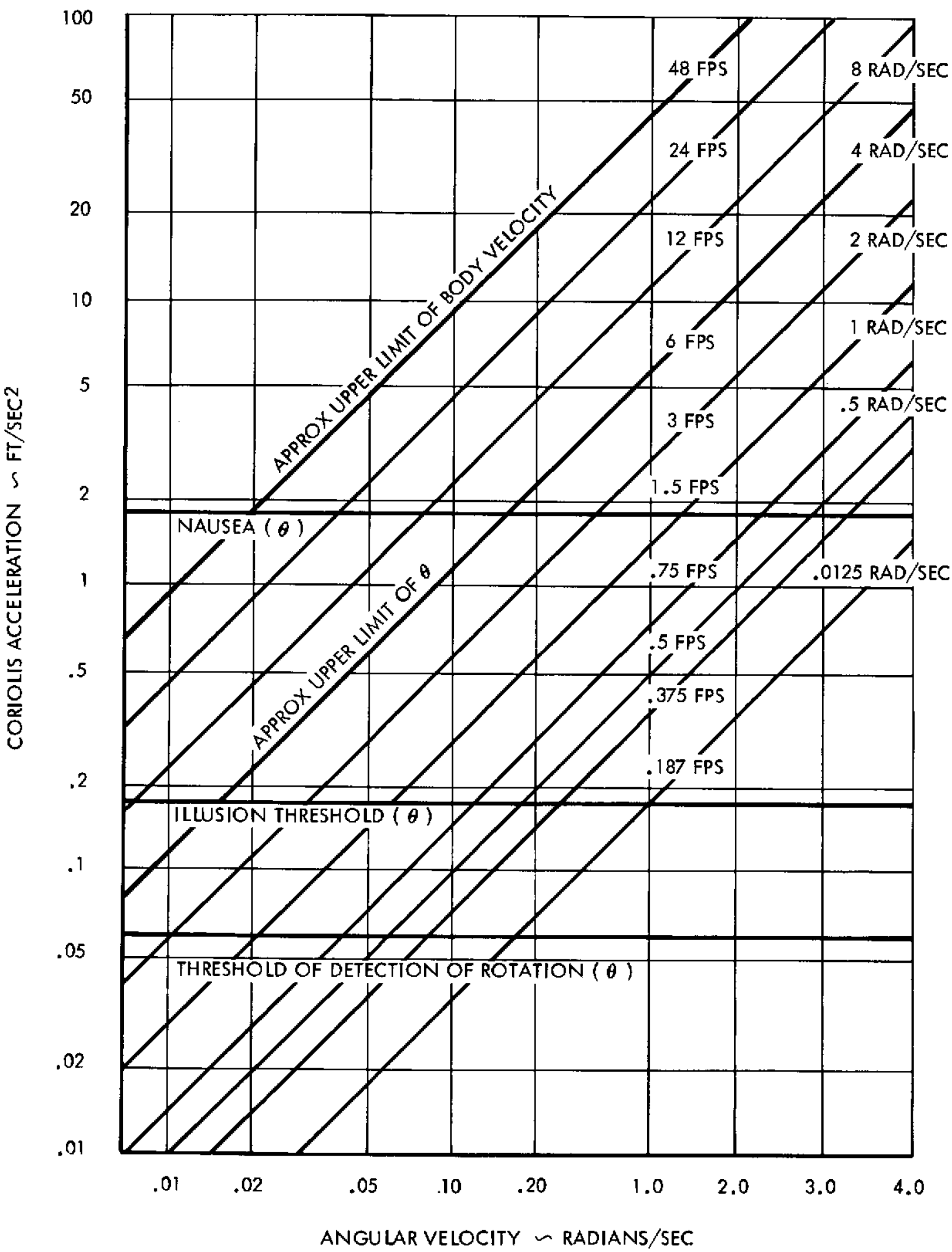


Figure 114. Angular Velocity - Radians/Second

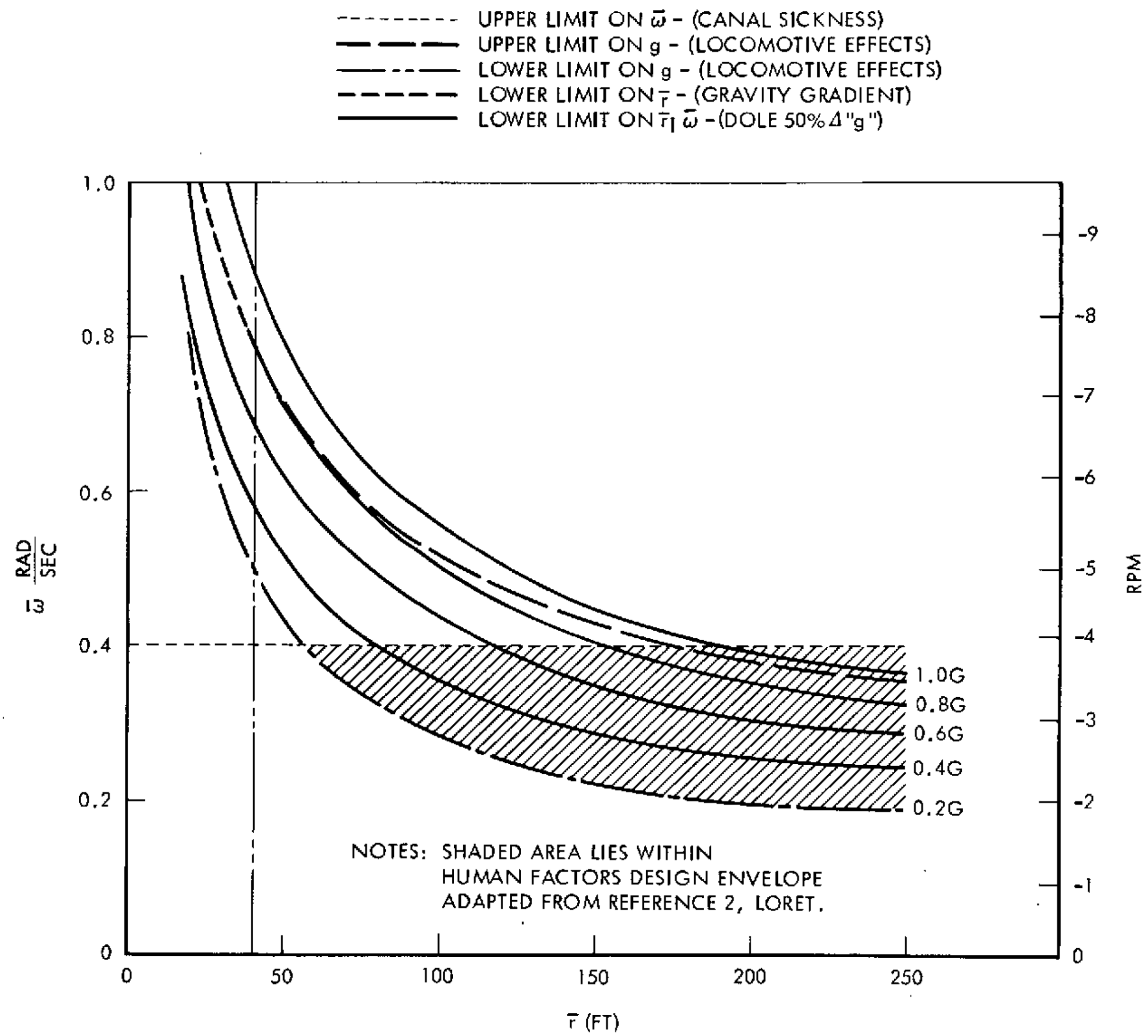


Figure 115. Human Factors Design Limits

4. The higher the rates of head rotation and of centrifuge rotation, the longer was the adaptation time. All test subjects adapted well at 1.71, 2.21 and 3.82 rpm, two of the five subjects had no difficulty in adapting at 5.44 rpm and none of those experiencing 10 rpm adapted completely.

5. After adaptation, the subjects used normal head movements (5 rad/sec) at 3.82 rpm but showed a tendency toward limited head movements at 5.44 rpm. Thus, there appears to be a crossover point or artificial boundary somewhere between 4 or 5 rpm.

Experimental evidence indicates that the rotational frequency of 2 rpm appears to be the threshold for a variety of physiological disturbances normally referred to as canal sickness. Adaptation to rotational frequencies above threshold does occur, after which task performance returns to that achieved under normal conditions. The time required for adaptation to take place varies directly with the frequency of rotation while the degree of adaptation shows an inverse relationship. Various experiments performed at the U.S. Naval School of Aviation Medicine, Pensacola, Florida, and the Naval Air Test Center, Johnsville, Pennsylvania, as well as supportive evidence from the USAF School of Aerospace Medicine, indicate that a rotational frequency of 3 to 4 rpm can be tolerated. However, the experimental research was limited to slow rotating room studies of limited duration (48 - 64 hours). At the suggestion of S&ID, a slow-rotating room study was conducted at the Naval Aerospace Medical Laboratory to determine the long term effects of continuous 3 rpm on human performance. The results of these studies are similar to those of previous studies conducted for shorter time periods in that the subjects adapted well at 3 rpm and apparently exhibited no additional problems.

Other experiments indicate that subjective reactions to rotation are also affected by the radius of the rotating vehicle. It appears that adaptation occurs at a faster rate at greater radii of rotation. This finding tends to strengthen the case for applying the data from the rotating room experiments to the manned space station environment. The data shows that a given angular acceleration is easier to tolerate with increasing rotational radii. With this fact in mind, the results of the recently completed study on the Naval Aerospace Medical Laboratory centrifuge, where subjects lived for 2 weeks in the rotating room at an angular velocity of 3 rpm without any apparent problem, give supportive evidence to the recommendation of this rotational speed in the design of the space station.

Visual Considerations

The preceding discussion has emphasized that people can react unfavorably to certain environments. One solution is to specify an environment that seems correct with respect to current knowledge. Figure 115 is a summary of the most recent analysis of force environment criteria. However, the briefest examination would show that the designer would have many questions left unanswered if the force environment was given as the sole human factors criterion.

Reference was made to canal sickness, illusions, and adaptation problems with respect to rotating stations. A brief review of the motion sickness literature will make it very evident that motion alone does not explain all facets of the sickness. For example, it is known that fewer passengers become airsick at night than in the day because external vision is reduced. In addition, many people become ill when they operate fixed-base, helicoptor simulators that include external vision simulation over wide angles. These experiences from everyday life and carefully controlled laboratory experiments all indicate that there is a strong interaction between motion and visual stimuli. It should be noted that prevention of motion sickness would not be adequate if the performance of the space station crew was adversely affected by the environment. The following discussion relates the salient facts from the visual and vestibular research literature to the design of the space laboratory.

Orientation

It is clear that one of the major causes of disorientation and the resulting dizziness, sickness, and decline in performance capabilities is a conflict between the visual, kinesthetic, and vestibular inputs. When people have a fixed reference, such as pilots have with the horizon, force environments which cause discomfort without this reference are not found to be as effective. The experiments in the U.S. Naval Pensacola rotating room, which were previously discussed have shown that much of the adaptation to rotational environment is lost when the door is opened and eyes can see the spinning of the room. In addition, the illusions of motion, both visual and bodily, can be induced by real or apparent motion. Many people have experienced the departure syndrome of motion after they have landed from a long ocean voyage.

Several design features can be included in the station design to control the incidence of visual/gravity conflict. A sense of vertical can be given by aligning interior walls along the radius vector. It is evident that reasonably small visual rooms are desirable to prevent the walls from appearing very tilted. Direct view of the exterior should be avoided. Mirrors and periscopes could be used which rotate in a manner which cancels station rotation if external vision is needed at the rim. Such vision would be desirable from a psychological standpoint if for no other reason.

Internal orientation, within the cabin vision, and the force environment all interact to determine either the acceptability of the design or the time required to adapt to the situation. It was shown in a preceding section that radii of at least 50 or 60 feet are required to provide a suitable artificial gravity environment. For purposes of visual/bodily orientation compatibility it is also necessary to consider the station radius.

At small station radii, a person standing at one end of a cabin on a curved floor would see the opposite wall as uphill and tilted toward his head. If he turned around and faced the near wall it could appear to be falling in on him or he would feel as if he were falling backwards. As the diameter increases, the angles between floors, walls, and ceilings can be reduced. The distorted rooms at amusement parks are good examples of the sensations which can be evoked when the visual and gravity orientations are in conflict. Studies by Witkin (References 3 and 4) are representative of research applicable to orientation criteria. The distortion between body tilt and wall tilt should be kept as small as possible; 8 degrees is a probable minimum upper limit and 3 degrees is the recommended value. A cabin layout, using 3 degrees will be found in the design section. Figure 116 shows the room lengths which could be used as a function of station radius for various wall angles.

Safety

If the self-deploying space station is to fulfill its potential as a scientific laboratory, as a staging base, or as a base for other functions that would require the number of personnel it is capable of supporting, safety becomes a predominate concern. A number of the design features that contribute to the inherent safety of the station are discussed in the design section. This section describes the advantages and disadvantages of the current concept with respect to the human factors aspects of safety.

Hazard Analysis

The major critical conditions and environmental events that would constitute an emergency condition are shown in Table 36 along with their probable remedial measures. For certain applications or designs, other types of subsystem failures could be classed as emergencies in addition to those shown in Table 36. For example, complicated experimental apparatus would fail and the functional aspects of the mission would have to be terminated. However, these latter items would probably not constitute hazards to the crew unless control was lost of some hazardous type experiment.

Radiation

It is well known that a major solar radiation event can produce fatal levels of radiation. If techniques can be developed to predict these events,

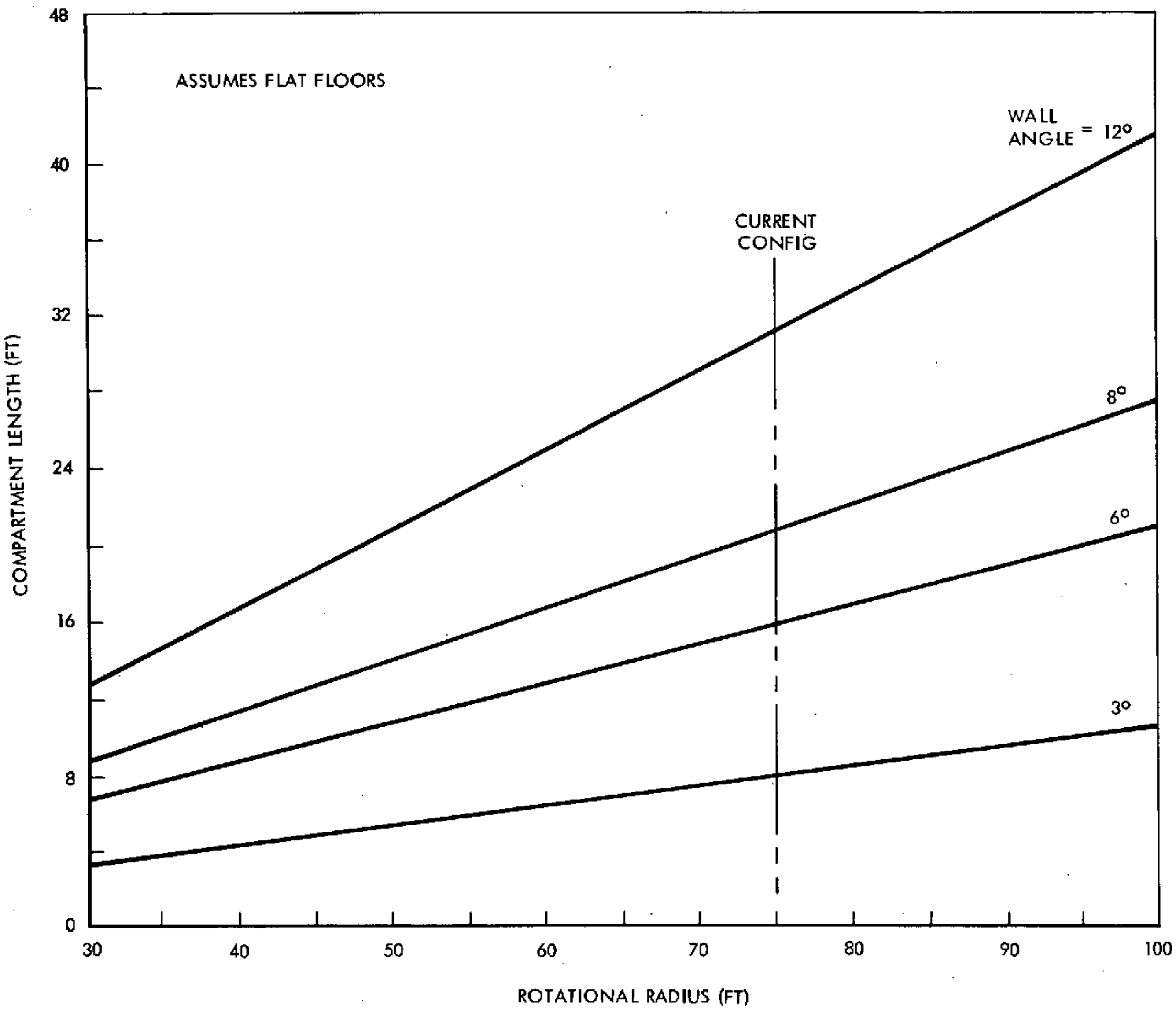


Figure 116. Compartment Length Considerations

Table 36. Orbital Emergency Conditions and Remedial Measures

Remedial Measure	Radiation	Puncture (Major)	Puncture (Minor)	Fire	Environmental Control System Loss	Spin Control Malfunction	Wobble Damper Failure	Electrical Power Failure	Communications Failure	Toxic Contamination	Major Explosion (Minor Treated Like Fire)
Escape	X	X		X	X		X	X		X	
Cabin Dump				X						X	
Pressure Suit		X	X	X	X			X		X	
Repair			X	X	X	X	X	X	X	X	
Standby Equipment				X	X	X	X	X	X		
Medication	X			X	X					X	

the station personnel can return to earth in the Apollo vehicles. As the accuracy or time of prediction decreases, rapid escape must be considered as a probable requirement. The only alternatives to escape are shielding and medication.

It is beyond the scope of this report to determine specific shielding requirements. However, it should be noted that any shielding that can reduce the cumulative dose from all sources of radiation also improves the operational economy of the space station. For example, the number of different crews required to continually man the space station will be a partial function of the maximum permissible radiation dose that may be received while in orbit. If the station is in certain orbits during periods of moderate solar activity, the tour of duty must be such to prevent an excessive exposure to radiation. If the duration of this duty cycle is reduced because of radiation, the personnel logistics cost increases. The additional expense of shielding could be more than justified by the duration increase in tours of duty that would reduce launch rate. Cost effectiveness studies should be conducted relative to these considerations.

Medication, which reduces or eliminates the biological damage of ionizing radiation, is theoretically possible. The radioprotective agents have generally been found to be toxic to the higher-level mammals. However, S&ID and other agencies are investigating certain combinations of chemicals that will provide protection without the toxic effects. Recent S&ID experiments show that these combinations can be tolerated by laboratory animals and can also provide effective radioprotection.

Punctures

Punctures could be caused by meteoroids, by collision with another spacecraft, explosions, or by equipment failures. The use of separate environmental control systems for each module, the use of separate airlocks, and the availability of the Apollos at dock greatly reduce the fatality potential of the station. The large volumes and adequate pressure levels recommended for the station compartments will result in a relatively long period of time before air loss is great enough to cause injury.

If two or more compartments receive major damage simultaneously or in close sequence, it is recommended that the mission be aborted. It may be possible to delay actual escape, but continued occupation of the station would probably not be safe. This is especially true if the damaged modules are those which contain the access tubes to the hub.

Fire and Toxic Contamination

The remedial measures for fire or toxic contamination are very similar in an orbital vehicle. In that separate atmospheric supplies are provided for each cabin, dumping cabin air can be an effective means of combating fire or eliminating contaminants. Even without resupply, the cabins could remain useful by space-suited personnel. The fire hazard is also effectively reduced by the use of an atmospheric composition which contains a significant percentage of diluent. The 10 psi pressure recommendation eliminates the fire hazards associated with pure oxygen, as would be required at low cabin pressures. Similar considerations can be seen in relation to contaminant production and buildup.

Equipment Failure

Hazards associated with equipment can usually be avoided by the use of redundant systems and by designing for maintainability. The current self-deploying space station design concept would be rated very high with respect to safety design as it relates to equipment.

Safety Problems

The problems discussed relative to the artificial gravity environment constitute problems with respect to safety. It was mentioned that a period of adaptation would have to occur before station personnel would be at full capability when entering or leaving the nonrotating hub. Emergencies which required rapid evacuation of the station and return to earth would be difficult to accomplish due to the probable adjustment period required. It is likely the various Apollo crews would be dizzy, nauseous, or ill for up to several hours after departure from the station. This is obviously a risky condition if reentry must be initiated as fast as possible.

Traffic problems that would occur during rapid escape conditions are also considered in the current design concept. Some form of transporter, such as the endless belt, is very desirable for rim-to-hub transport because of the unusual forces during radial motion.

In general, any vehicle from which transfer of personnel to other vehicles is required during emergencies is less safe than a vehicle capable of reentry. However, the inherent safety of the multicabin design with redundant life support critical systems, greatly reduces the frequency of occasions whicy would require escape. In addition, the problem of disorientation may be relieved by drugs, training, and selection.

RESEARCH AND OPERATIONAL APPLICATIONS

One of the predominant justifications of the self-deploying space station concept, from the human factors standpoint, is that the system itself is required to solve many of the research problems. The first research application of the space station will include an evaluation of the adaptation of the crews to the station. The rational requirement for this type of vehicle cannot be fully validated without the vehicle.

The rotational velocity of the station can be varied by the crew if desired. Actual optimization of the rotational velocity for the final radius can be done in the first launch using the Apollo crew. Many of the theoretical problems and recommendations associated with the force environment cannot be precisely extrapolated to a reduced gravity situation. The many advantages of artificial gravity cannot be realized unless an experimental system is developed which explores this factor.

Attempts to simulate gravity by small vehicles connected by cables or the like may give negative results due to interactions between the various sensory receptors, the confinement stress, and other factors that contribute to complex reaction to rotation. The self-deploying space station concept would not necessarily be invalidated if early artificial gravity experiments fail.

TRAINING

The training required for personnel who will man the station must be adequately planned and completed in advance of the required operational date. It is possible to offer a tentative program at this time and to indicate areas which must receive subsequent study and research effort.

It should be noted that an evolution and probably a simplification of training will occur as additional space stations are launched or as crews are replaced. The first crew should be an Apollo qualified crew. These personnel will have had extensive training in space flights, biological sciences, spacecraft engineering, etc. However, training will be required to accustom crew members to rotation and to familiarize them with sensations associated with this type of environment. Emergency procedures training will be required and will be integrated with the procedural training needed to effectively operate and repair the space station systems.

Therefore, three training areas are indicated, experimental, emergency, and procedural. The procedural training should include thorough training in the engineering and design of all aspects of the space station with different crew members receiving emphasis relative to prior skills. This type of program would probably take from one to two years with the latter portions requiring full-time participation.

As the operational problems of the vehicle become known and the various subsystems are debugged, the required training can be reduced.

The various scientific specialists would require training in only the emergency procedures and experiential training to the environment. However, it is likely these people would not be Apollo qualified and would, therefore, require selected Apollo training. Subsequent studies or procurement effort must give adequate attention to a detailed investigation of the required training program.

LOGISTICS

In manned space operations, reliability and maintainability are fundamental to survival. These requirements must be integrated and treated jointly because they are interrelated; decisions affecting one will usually affect the other. Analyses of the mission of the space station indicate a need for improved methods for predicting spares requirements, particularly when considering that, historically, replacement parts have uncertain demand rates and widely varying characteristics. These are only a minor part of the system support requirements that are required to maintain men and equipment in the space environment for extended periods of time. Replenishment of gas and water lost through leakage and system inefficiency and replacement of food and other consumable items make up the bulk of the weight and volumetric requirements for the resupply mission.

Since extensive analyses of operational techniques were not conducted, several factors have been assumed. These include the assumption that rendezvous techniques have progressed to a degree that no major problems are anticipated, that the crew build-up will be gradual to a total of 21 men, and that the space station will be in full operation at the end of 1 year with no phase-out of personnel. At that time, all on-station personnel will be returned to earth or operations will continue depending on the integrity of the system and a re-evaluation of the program.

Three modes of meeting the support requirements have been considered:

1. Use of the Apollo command module with a resupply module inserted between the command and service module or aft of the service module

2. Increase of the personnel capacity of the command module to five and interspersing personnel and resupply missions

3. Retention of the Apollo in its earth orbital configuration and removing unnecessary equipment from the command and service modules to provide additional weight and volumetric capacity.

Of these three modes, the third most nearly satisfies the requirements of the space station operation. The Apollo vehicle, in its earth orbital configuration, provides the capability of carrying the crew and approximately 2200 pounds of equipment without degrading the abort system. Volume remains a problem that may be solved through the use of optimized package size and the construction of special on-board handling racks for weight distribution and efficient storage. Space within the module appears to be no problem considering the fact that the command module will be operative for less than 24 hours in each direction (i. e., earth to orbit, and orbit to earth).

The present stage of conceptual design precludes a detailed analysis of the factors that define support requirements. Therefore, past government and industry programs were examined in this study to establish a base from which to predict support requirements. Where similar systems were used, the firmest data were extrapolated and used to establish gross requirements. The resultant procedures and requirements are predicated on established reliability goals, maintainability (service and repair), and usage rates of consumables.

Package size is influenced by a number of factors, the ability to handle airlock size, and storage compatibility being prime. The size of the airlocks in the space station and Apollo module, and the position of the individual in traversing these locks indicate that a package approximately 2 cubic feet in volume approaches the upper limits of package size. This particular constraint, coupled with a maximum gross earth weight of about 100 pounds, will provide optimum handling characteristics in the shirt-sleeve environment of the space station at the hub and in the modules.

DETERMINANTS

The basic determinants of support requirements are maintenance, usage rates of consumables, the effectiveness of the life support system, and crew rotation. The selection of items for on-board stowage and subsequent resupply missions reflect their importance to mission accomplishment. Highest priority has been awarded to those items which affect crew safety and survival, such as navigation spares, life support equipment, station repair materials, and supplemental food and water supplies. Scientific and data-gathering equipment and items that provide added measures of reliability, confidence, and comfort are secondary in nature.

Consumables comprise approximately 90 percent of the gross resupply requirements in weight and volume, as indicated in Table 37. These elements are divided into two basic categories; (1) man's requirements other than those provided by the environmental control and utility and waste systems and (2) losses through uncontrolled escape. In the first category, food is the major item. Previous studies indicate that the average man on a space mission will consume from 1-1/4 to 1-3/4 pounds of food per day, depending on the degree of dehydration. In connection with the Apollo program, a figure of 1.82 pounds of packaged dried food was established as

Table 37. Annual Resupply Requirements (Excluding Crew)

Item	Planning Factors	Estimated Spec. Grav	Remarks	Weight (lb)	Volume (cu ft)
Dry Food	66 lb/man/year (packaged)	0.5		13,986	448
Water	100 lb/6 weeks	1.0		866	13.9
Water Containers	2 Gallon Plastic	—	Assumed 1% of water volume and 1 lb per container	50	1.4
LO_2	1.32 lb/day			482	Incl in containers
LN_2	5.68 lb/day			2,070	Incl in containers
LO_2 and LN_2 Containers	44 Containers per year		Vacuum jacket stainless steel container weighing 36 lb and displacing 1.57 cubic feet	1,583	69.0
Consumables	1/2 lb/man/day	0.3	Tissue, underclothing, etc.	3,840	205
Scientific Consumables	0.5 lb/day	1.0	Film, tapes, etc.	183	3.0
Contingencies	1000 lb/year	1.0	Unscheduled spares, etc.	1,000	16.0
Storage Loss			Storage volume loss due to cylindrical shape of gas containers	—	50.0
			Totals	24,060	806.3

a planning factor utilizing existing processing and packaging techniques. This figure establishes 666 pounds of packaged dried food per year for each man, or a gross requirement of about 14,000 pounds for 21 men.

In computing these requirements, phase-out at the end of 1 year is not considered. If phase-out is desired, the requirement would be reduced proportionately for those crew members whose tour on station would be less than a normal resupply cycle. This would also be true of other consumables.

Daily water ingestion will average about 5 pounds per man. Additional water will be required for hygienic purposes. The effectiveness of the environmental control and the utility and water waste systems will determine specific water resupply requirements. Almost total water reclamation has been assumed; therefore, only replacement of water lost through uncontrollable escape is considered. This would be in conjunction with gas leakage or severe contamination. The lack of verified data has resulted in establishing an assumed requirement of 100 pounds of water for each 6 weeks of operation.

Sanitation requirements for cleaning of clothing and similar applications have been considered and are assumed to be unmeasurable or negligible. The use of materials made from polyethelene terephthalate (Dacron) and similar plastics are conducive to cleaning by a degassing process which makes use of the vacuum conditions in space. Clothing, blankets, sheets, and similar articles may be placed in an airlock and exposed until clean. This concept reduces initial lift-off weights and eliminates separation of detergents from water.

Oral, personal, and group hygiene dictate the use of varied consumables such as tooth paste, soap, toilet and facial tissues, underclothing, and other items of a similar nature. Best estimates from Apollo and similar studies indicate that an average daily weight requirement of 1/2 pound per man will satisfy these needs.

Oxygen and nitrogen replacement rates are based on the probable gas leakage rate of 10 percent per month.

The specific scientific functions of the laboratory are not clearly defined at this time. However, it is assumed that the function will generate requirements for chemicals, data recording materials, and similar functional devices utilizing expendable materials. The usage rate of these materials cannot be fixed at the present time, but will depend on the rate of scientific data flow and environmental damage. In the absence of specific data as to function and type of equipment, a factor of 1/2 pound per day has been assumed.

Although significant advances are being achieved in equipment reliability, the present and projected state of the art is such that replacement and/or repair is necessary in the case of equipment operating over extended time periods. Components must be replaced for varying reasons including damage, wear, and predicted service life. Random failures can be expected. All failures affecting mission accomplishment must be rectified either through replacement, if the required component is available or by repair. To provide for contingencies of this nature, 1000 pounds per year over the basic spares provisioning, have been allocated and are included in Table 37.

The quantities defined here are based on usage factors and other determinants. It also is essential that basic quantities be defined. The lack of definitive design data precludes the use of specific reliability values in determining these requirements. Table 38 contains estimates of requirements that are programmed for the initial launch. These estimates consider data from this and other studies that have progressed to the hardware definition stage.

The gross supply and resupply requirements have been divided into two parts; the loading for the initial launch and the material allocated to resupply missions. Figure 117 provides a means of equating space and weight available in the Apollo resupply vehicle to the number of launches required per year.

During the course of 1 year of operation, the accumulation of processed fecal material, tissue, discarded wrappings, and replaced nonreparable spares resulting from operation of the station equipment, is considerable. This waste material may be retained, discarded into space, or returned to earth via crew rotation missions. Retention serves no useful purpose, particularly in view of the limited storage space aboard the space station. Indiscriminate discarding into space would add to the accumulation space debris. In the absence of established policy regarding waste disposal in space, return to earth appears to be the most feasible procedure. Table 39 contains estimates of waste material which must be considered regardless of the method selected. It appears that the quantities involved fall within the heat/weight limits of an Apollo-type vehicle.

OPERATIONAL CONSIDERATIONS

The resupply launch frequency and use of modified Apollo modules as cargo carriers, have been established. However, despite the amount of reliability data available by the time the system becomes operational, it is probable that random failures may occur which have not been provisioned for and which could threaten crew safety or impair accomplishment of the

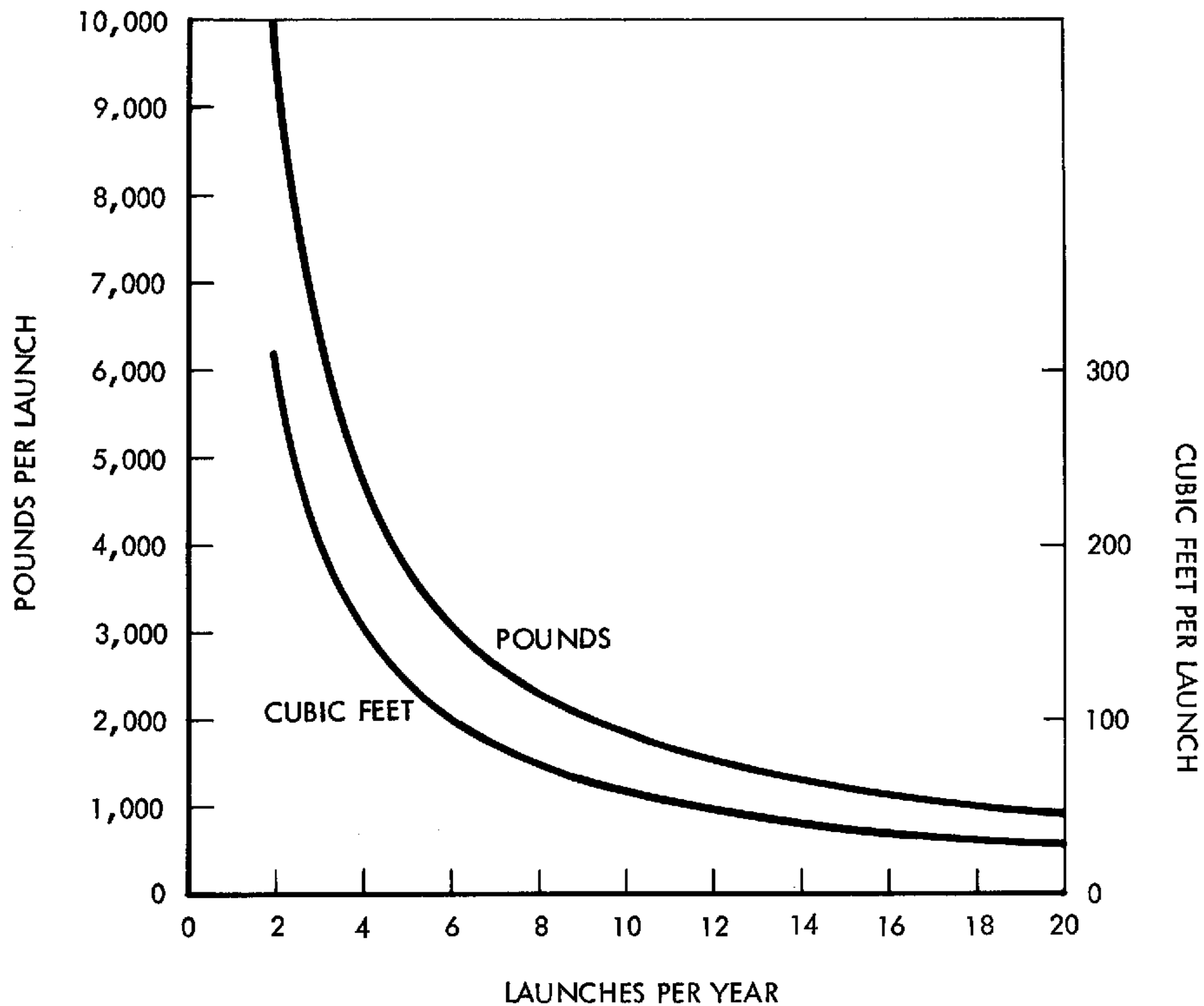


Figure 117. Annual Resupply Requirements (Excluding Crew Rotation)

Table 38. Basic Launch Supply Requirements (Excluding Crew)

Item	Planning Factors	Estimated Spec. Grav	Remarks	Weight (lb)	Volume (cu ft)
Bedding	18 sets		2 lb/set	36	2.6
Communications (600 lb)	10 percent of weight per year	1.0		60	1.0
Scientific Equipment (8,320 lb)	7 percent of weight per year	1.0		583	9.4
Maneuvering Fuel	One year supply			3,000	
Tools and Test Equipment		1.0		225	3.6
Utility and Waste System			Filter life - 6 weeks Moving parts life - 6 months	160	4.8
ECS			Filter life - 6 weeks Moving parts life - 6 months	65	1.3
Water	7 lb/man/day 3 day's supply			3,640	58.8
Medical Supplies			1 kit per module and hub	105	2.6
Structural Repair Kits			1 kit per module and hub	105	2.6
Emergency stores			3 kits per module and hub (Food kits and O_2 bottle)	420	10.5
			Totals	8,399	97.2

NOTE 1. In addition to the above initial loading, provision should be made for 1/N times the resupply requirements contained in Table 37, where N = number of launches per year.

NOTE 2. Weight and volume figures include packaging.

Table 39. Estimated Refuse Disposal Requirements (Annual)

Item	Planning Factors	Estimated Spec. Grav	Remarks	Weight (lb)	Volume (cu ft)
Fecal Waste	0.33 lb/man/day Less 75 percent H_2O		Assumed spec grav = 1.25	635	8.4
Misc. Waste	0.5 lb/man per day		Tissues, etc. - Assumed spec grav = 0.2	3,840	307
Spares and Food Wrapping			Assumed spec grav = 0.2	560	49
Water containers				50	14.3
Gas containers				1,583	69
			Totals	6,668	449.7

NOTE: Certain replaced modules may be added, depending on economics of repair versus discard.

scientific mission. For this reason, a quick reacting emergency resupply system embracing communications, materiel back-up, and unscheduled launch capability must be provided.

SAFETY

Safety and survival, in the face of catastrophic damage to one or more modules, are synonymous. First consideration is given to atmospheric requirements, in that without oxygen, man's life span is measurable in seconds. Lightweight flexible space suits and portable life support units must be available in each module in sufficient quantities to meet the needs of the regular inhabitants who normally operate in a shirt sleeve environment. The actions of the crew members would depend on the seriousness of the damage and/or personnel casualties. It is essential that an emergency doctrine be established so that crew members may take effective steps to preserve life and equipment. Provision has been made in the basic loading plan for medical and repair kits and emergency rations. Additional study will be required to develop optimum procedures. These, when integrated into the training program through a series of scheduled and unscheduled emergency drills, will result in the desired proficiency.

MAINTENANCE

The concept that reliability and maintenance, including replacement and repair, are interrelated and that changes in one induce changes in the other bears heavily on maintenance concepts and philosophies. Repeated attempts have been made to measure maintainability (i.e., the capability of being retained in or restored to serviceable condition). As of this time, fixed factors have not been established. Based on various studies in this field, maintenance time, defined as the total number of man-hours of direct maintenance expended, is used as a criterion to establish maintainability. In the space station, the maintenance factors coupled with replacement factors become critical in terms of time, weight, and space. Past experience has indicated that trouble isolation is a major time consumer. Efficient self-test features and design of other support equipment to increase speed, accuracy, and reliability of the maintenance operation will materially reduce maintenance times. In determining the degree to which self-test equipment will be utilized, the designer must consider weight and complexity factors, the effectiveness of other checkout equipment as compared to self-test, and the relative importance of the subsystem to survival and mission accomplishment.

Maintenance can neither depend on redundancy or automatic equipment to completely satisfy its needs. It is necessary to realistically face the fact that in-orbit maintenance will be necessary because of the probability of random failures as well as scheduled maintenance events.

The degradation of equipment effectiveness is a function of time as represented by a typical characteristic curve in Figure 118. This curve is a function of random and wearout tendency to fail. To establish a point in time where maintenance becomes imperative, the minimum and desired reliability levels are plotted, with the marginal operational zone falling between.

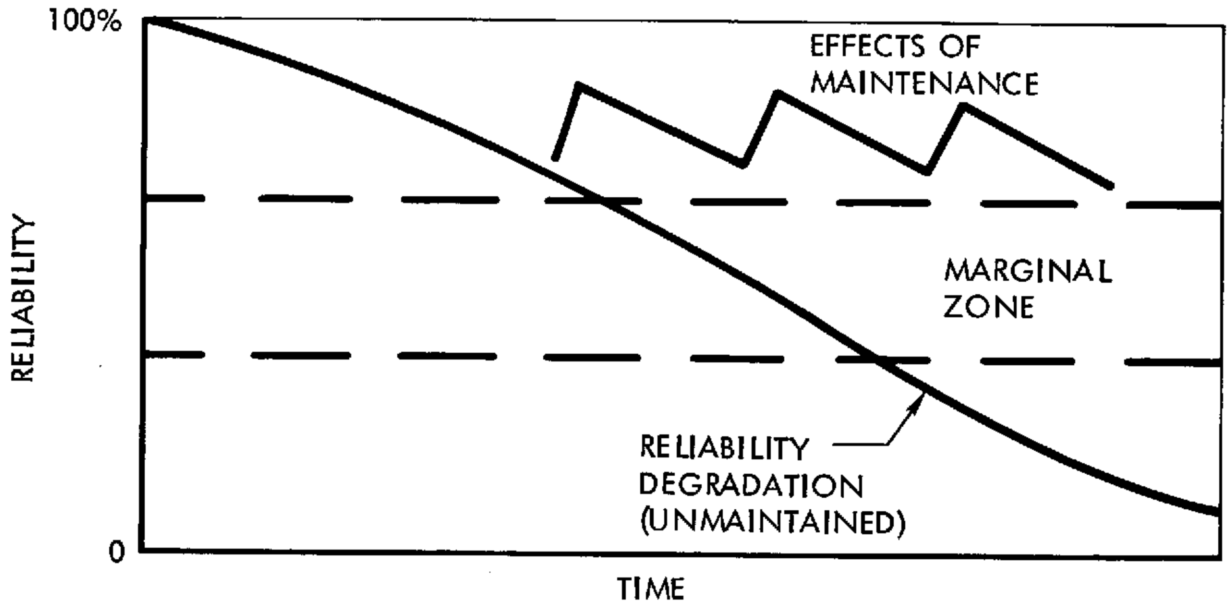


Figure 118. Degradation of Equipment Effectiveness

To maintain optimum operating characteristics during the mission life, it is evident that an improvement in component life factors is necessary. Assuming redesign has been fully exploited, the accepted methods of effecting improvement are redundancy and maintenance, including component replacement and servicing. The degree of redundancy required to provide the desired level of reliability can become prohibitive, particularly where long periods of operation are involved. Preventive maintenance, however, can raise the curve back to or near the original levels of reliability and becomes a most feasible method in the case of the space station.

Consideration has been given to the level to which maintenance can be reasonably and economically conducted. Except in rare instances, these considerations dictate modular replacement and limited repair capability augmented by essential redundancy. The trade-offs between environment, spares acquisition, training time, residue disposal and other similar support limitations will be the final governing factors in follow-on phases. Spares weight and cube factors established during this study are based on space availability and consideration of the foregoing.

OPERATIONS ANALYSIS

This section includes a discussion of some of the factors associated with the transfer of the space station crew members between the earth and the orbiting vehicle. Cost factors are presented as a function of mission duration and crew size. Other resupply functions (e.g., expendable materials), which are closely allied to the personnel transfer problem, have been discussed separately in the Logistics section of this report. Together, the two sections provide a basic definition of some of the major considerations in operating a large, multimanned space station over an extended period of time.

PRIMARY FACTORS

One of the major questions that arose during the course of the study was related to the size of the crew to be aboard the space station during normal operations. To firmly establish the minimum number of men required involves a very detailed analysis—beyond the scope of the present effort—of the tasks that must be performed. The maximum number of men is a function of the tasks, available volume, interior design, launch weight, and available docking facilities. With only the Apollo vehicle for use as a personnel transport, the operational costs, which are a function of mission duration, also become a very important parameter in establishing crew size. While time and funds for the basic study effort prevented the required detailed task analysis, a limited study of the effects of operational cost factors on the crew size was conducted.

One of the assumptions employed throughout the study was that existing spacecraft and launch vehicles would be utilized in the operation of the space station. This implies, of course, that the Saturn launch vehicle family be used and that one of the three existing or planned NASA reentry vehicles—Mercury, Gemini, or Apollo—be used as the personnel transport and resupply vehicles. In order to minimize the number of launches and rendezvous maneuvers required, it becomes desirable to use the vehicle that carries the largest number of crew members and resupply items. The Apollo appears to be the most suitable, if not the only acceptable, vehicle because neither Mercury nor Gemini have an airlock required for the transfer to the space station; and, in addition, it has a capacity of transporting at least three men per trip. With the removal of equipment associated only with the Apollo lunar mission, it may be possible to increase the capacity to five or six men.

Figure 119 shows the relationship between the three inter-dependent variables—mission duration, crew size, and number of launches required for a 1-year operation. These curves are based on the use of an Apollo vehicle carrying three crew members per launch. This chart can be used in several different ways. For example, if a mission duration of 26 weeks is assumed, only four launches per year would be required to maintain a six-man space station crew, but 14 launches would be required for a 21-man crew size. Similarly, if it is desired to limit the number of launches per year to a given level, then the required mission duration to maintain a given crew size can be found.

It is currently believed that, at least in the early space station operations, the mission duration for the crew members will be limited to about six to eight weeks because of the unknowns associated with long operations in space. It can be seen from Figure 119 that, for short mission durations, the number of launches required increases very rapidly with increasing crew size. For a crew of 21 men, 63 launches (more than one per week) are required to maintain one station in orbit for a 1-year period.

In recognition of the large number of boosters required for these operations, it becomes desirable to investigate possible methods of carrying more crew members per launch. One possibly obvious approach is to increase, if possible, the number of men in the Apollo spacecraft; another is to utilize a transport vehicle which carries half or more of the space station crew members at one time. The effect of these alternate approaches on reduction of the required number of launches can be deduced from Figures 120 through 122.

Several factors must be considered in establishing an optimized approach to the personnel transport and resupply operations for the space station. The NASA Saturn C-1 on the Air Force Titan III can be used to launch either the basic three-man or a modified five-man Apollo. A reentry vehicle that carries half or all of the space station crew members will require the development of a new launch vehicle or else the use of the Saturn C-5 since no intermediate-payload booster is presently planned. The development costs of any new booster as well as those of a new personnel transport vehicle would need to be amortized over a given period of space station operations.

OPERATING COSTS

Preliminary cost studies that have been conducted have been devoted to an analysis of the use of the basic or a modified Apollo and to the use of a new multiman personnel and resupply vehicle. Cost figures that were developed early in the analysis and used in subsequent trade-off studies are presented in Table 40.

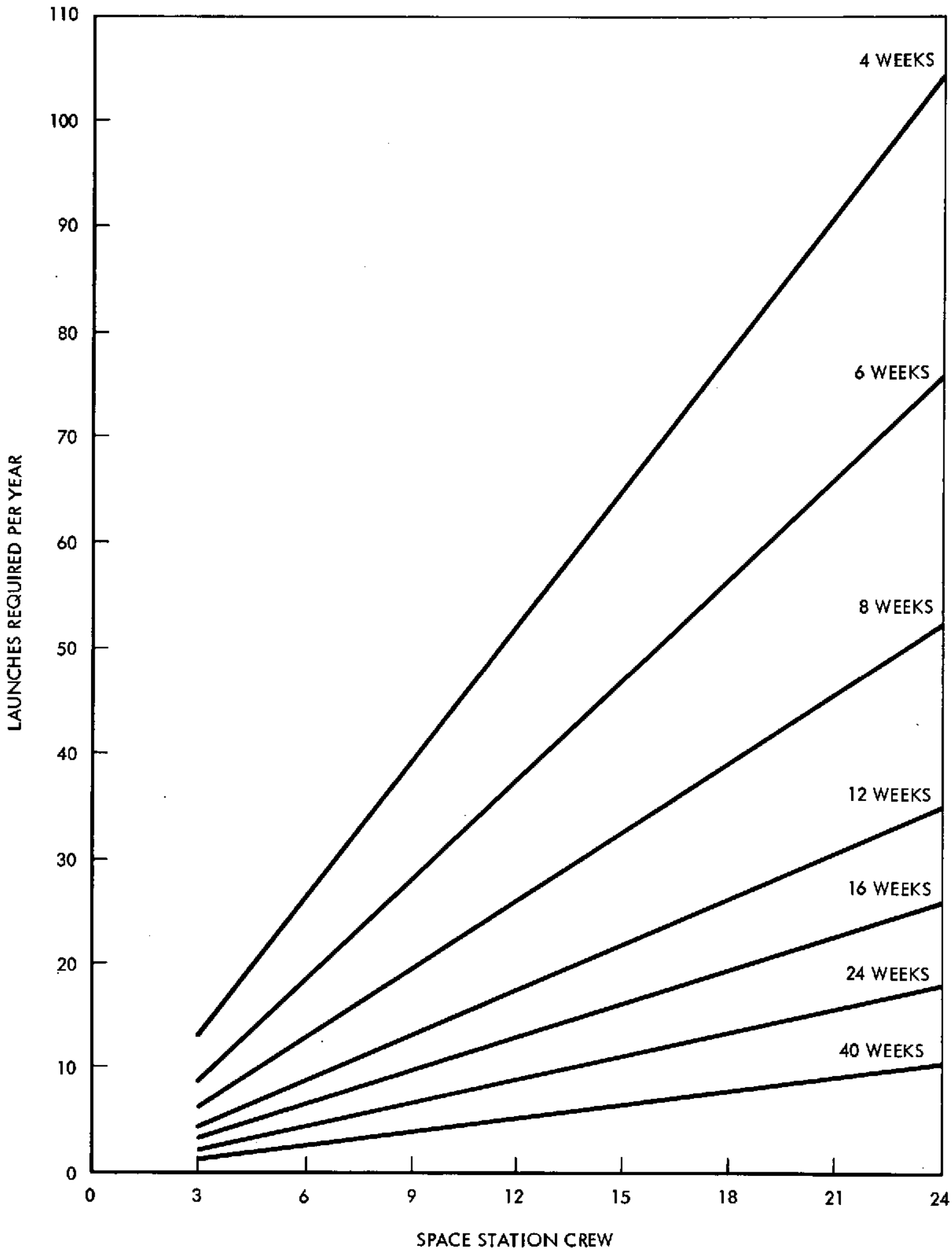


Figure 119. Apollo/Saturn C-1 Launch Requirements (3 Passengers per Launch)

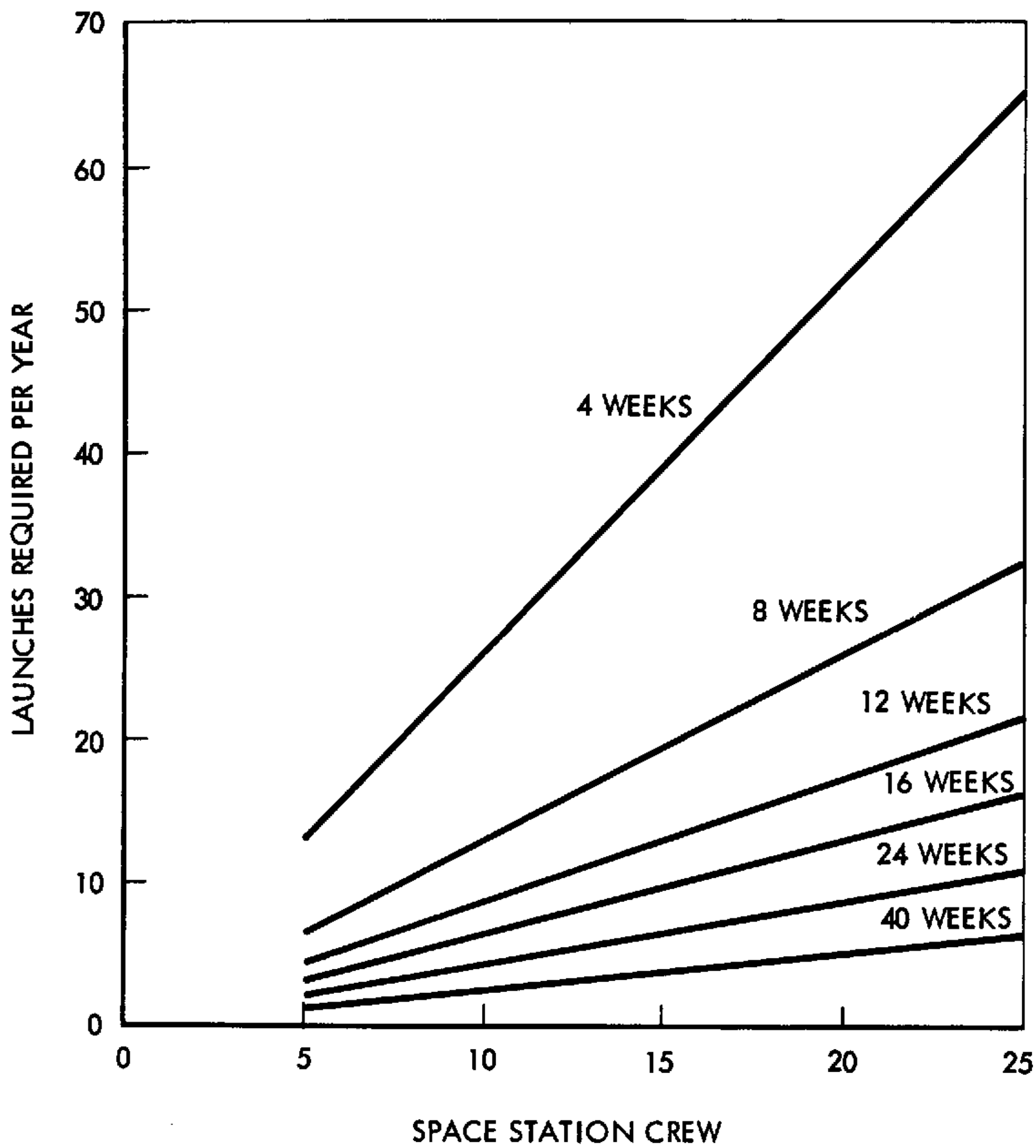


Figure 120. Apollo/Saturn C-1 Launch Requirements
(5 Passengers per Launch)

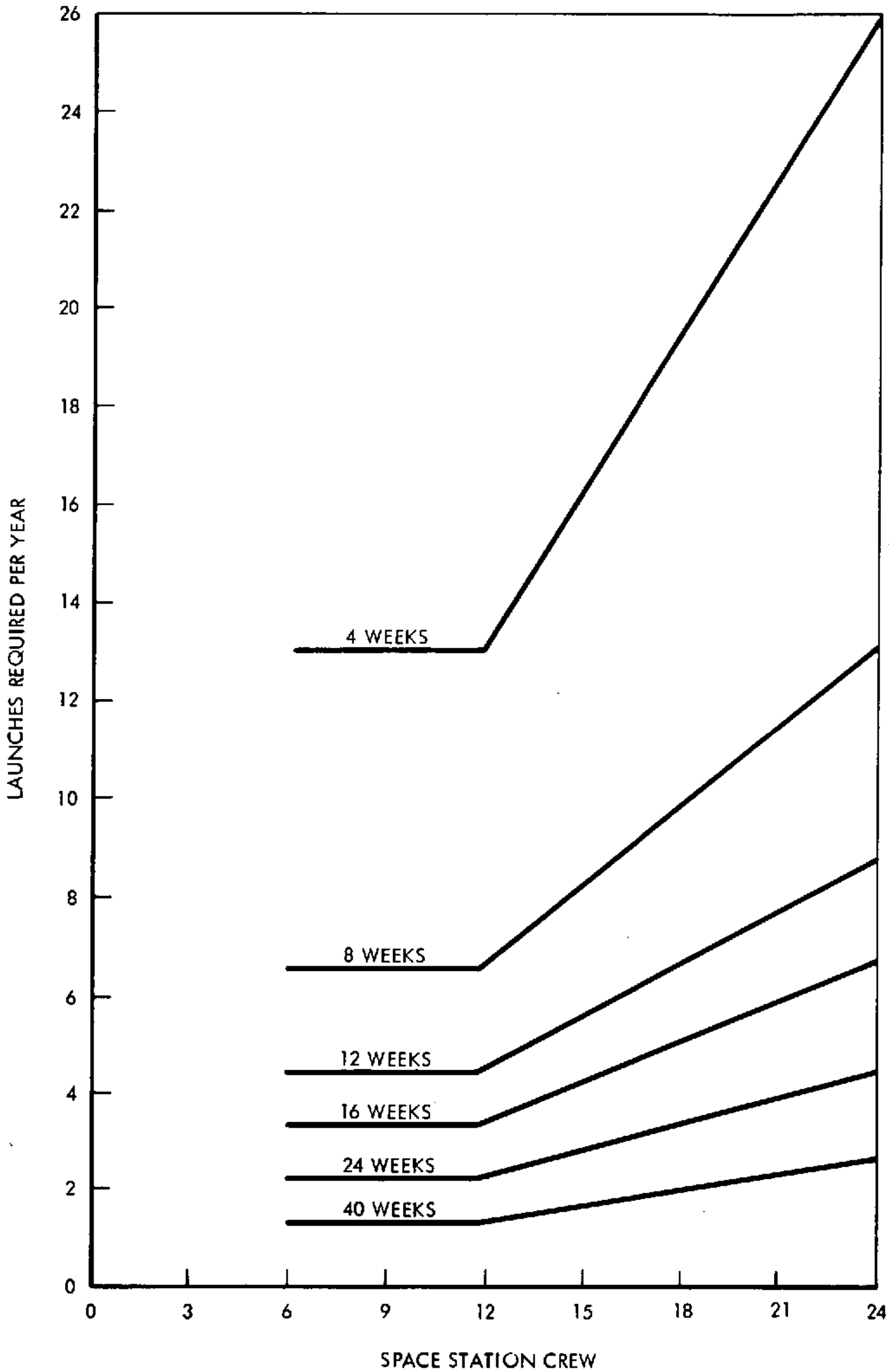


Figure 121. Personnel Transport/Saturn C-3 Launch Requirements (12 Passengers per Launch)

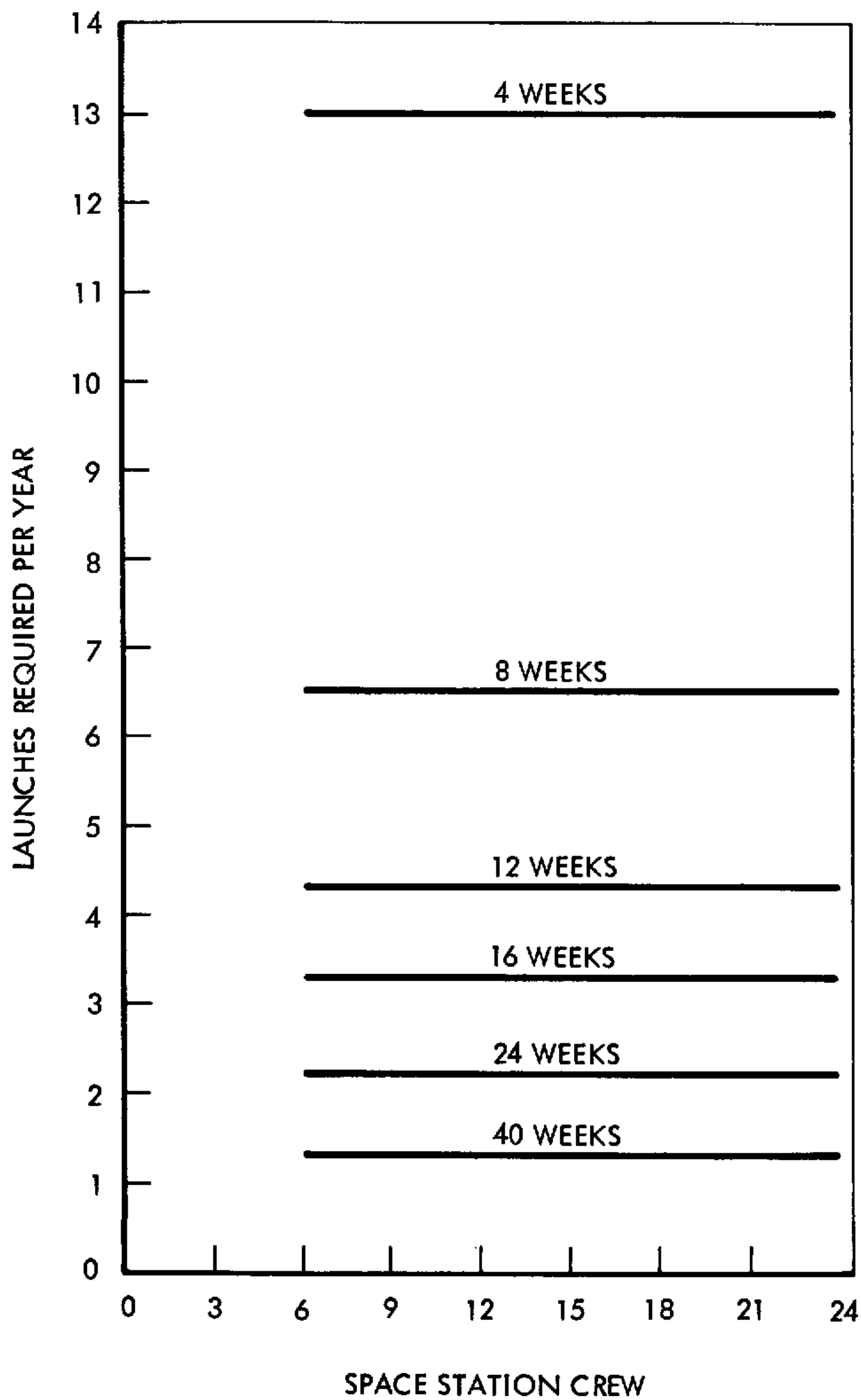


Figure 122. Personnel Transport/Saturn C-5 Launch Requirements (24 Passengers per Launch)

In determining the operational costs associated with use of the basic Apollo vehicle, it was assumed that the Apollo could carry three men plus the necessary resupply items for the particular mission duration under consideration. In this analysis, no allowance was made for loss of vehicles due to launch abort; however, it was assumed that each Apollo could be used only one time. Two different boosters, the Saturn C-1 and the Titan III were assumed for launching the Apollo. Figure 123 illustrates the operating costs (Apollo + Saturn C-1 boosters) for a 1-year period as a function of mission duration and crew size. In a similar study involving use of the Titan III booster instead of the Saturn C-1, it was assumed that the development costs of the former would be absorbed by the space station program. However, since the unit production cost of the Saturn C-1 is about one-third more than that of the Titan III, the net result is that a "crossover" occurs; and for more than 50 launches, the total program costs are actually less with the Titan III. Figure 124 shows a comparison of the one year operations costs for the two different boosters.

In another costing activity, a comparison was made of the space station operational costs for two basically different personnel transport vehicles, as well as for variations of these vehicles and the boosters used to launch them. By removing the equipment which is associated only with the lunar mission, the Apollo vehicle is assumed to be capable of transporting five men per launch. Furthermore, a 5-percent attrition rate on all launch vehicles is assumed; consequently the required number of personnel transport vehicles has been increased to reflect this loss. The Apollo vehicle was also assumed to be reusable one time with an associated refurbishment cost of 0.75 million dollars.

The other vehicle assumed is multiman personnel transport (lifting reentry vehicle) which can land at predesignated earth recovery sites. It is a configuration which employs hot radiative (rather than ablative) structure to enable it to be reused a sizeable number of times. Based upon separate S&ID design studies, a 24-man configuration would weigh about 50,000 pounds at launch. A 12-man configuration would be somewhat smaller (so as to be compatible with the Saturn C-1 or Titan III), but the development cost would essentially be equal for both vehicles (or has been so assumed). A refurbishment cost of 1 million dollars per mission has been used.

Having established the costs of the personnel transport and resupply operations for a 1-year period and having examined the cost trade-offs between the use of the Saturn C-1 and the Titan III, the analysis was extended to a 5-year operational period. The primary ground rules for this analysis were as follows:

1. Three self-deploying space stations will be in orbital operation at all times.

2. Twenty-one crew men will be continuously assigned to each station.

3. Duty cycle will be eight weeks for each crew member.

4. There will be a 5-percent attrition of vehicles, with uniform distribution over the 5-year period.

5. Initial launch costs for the three space stations are not included.

6. Production rates of various systems considered will suffice for launching rates. Spare or back-up systems are not accounted for.

7. Refurbishment costs are $0.75 million for the Apollo and $1.0 million for the multiman personnel transport.

The analyses for a 5-year period were based on the following manned vehicle and booster system combinations:

Case I - Apollo (three-man/no reuse) with Saturn C-1 booster.

Case Ia - Apollo (three-man/one reuse) with Saturn C-1 booster.

Case II - Apollo (five-man/no reuse) with Saturn C-1 booster.

Case IIa - Apollo (five-man/one reuse) with Saturn C-1 booster.

Case III - Multimanned Personnel Transport (NAA 21-man configuration/reusable) with Saturn C-5 booster.

Case IV - Multimanned Personnel Transport (NAA 21-man configuration/reusable) with Saturn C-3 booster.

Case V - Multimanned Personnel Transport (NAA 12-man configuration/reusable) with Saturn C-1 booster.

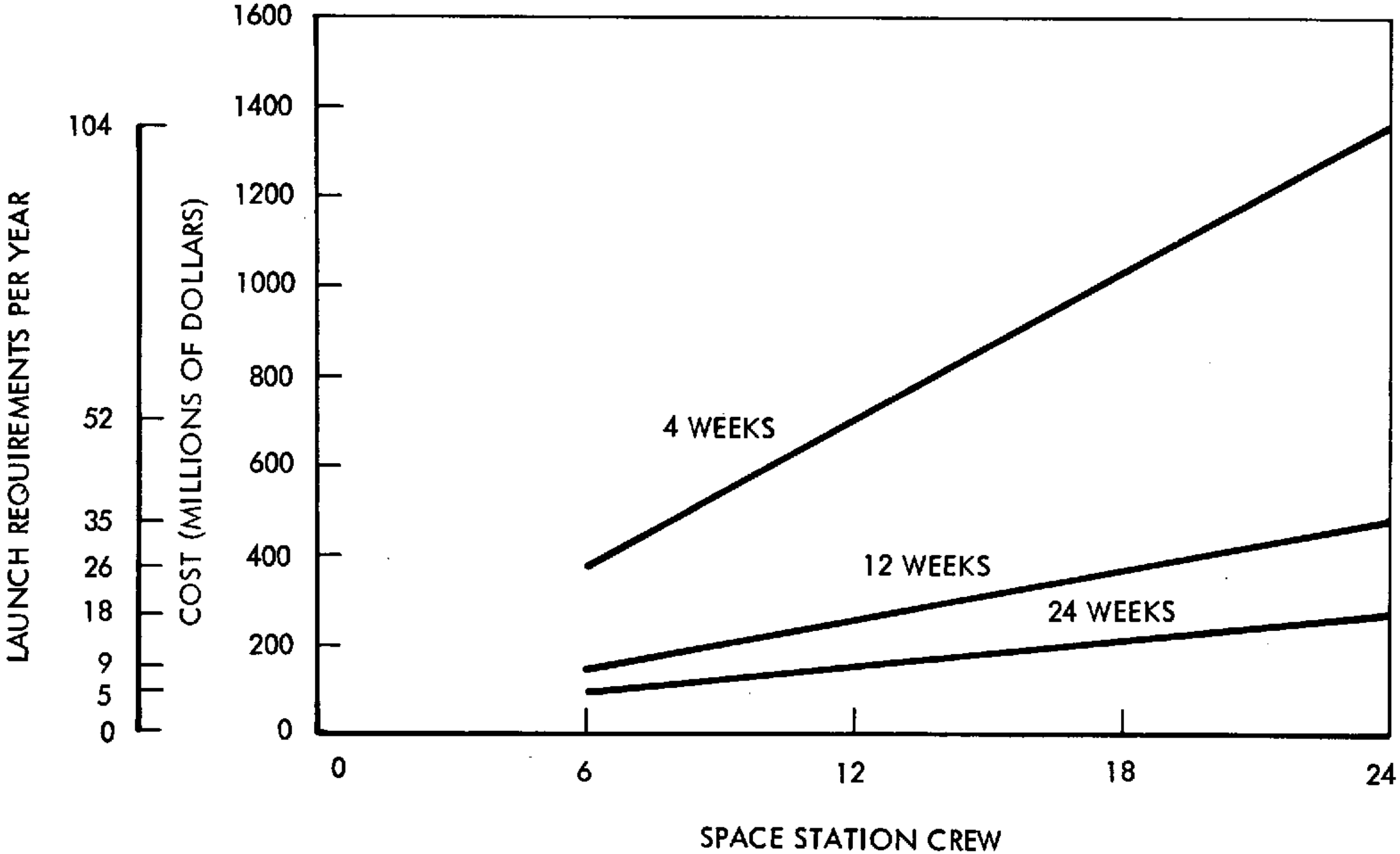


Figure 123. Total System Cost Versus Number of Space Station Crew (3-Man Apollo/Saturn C-1)

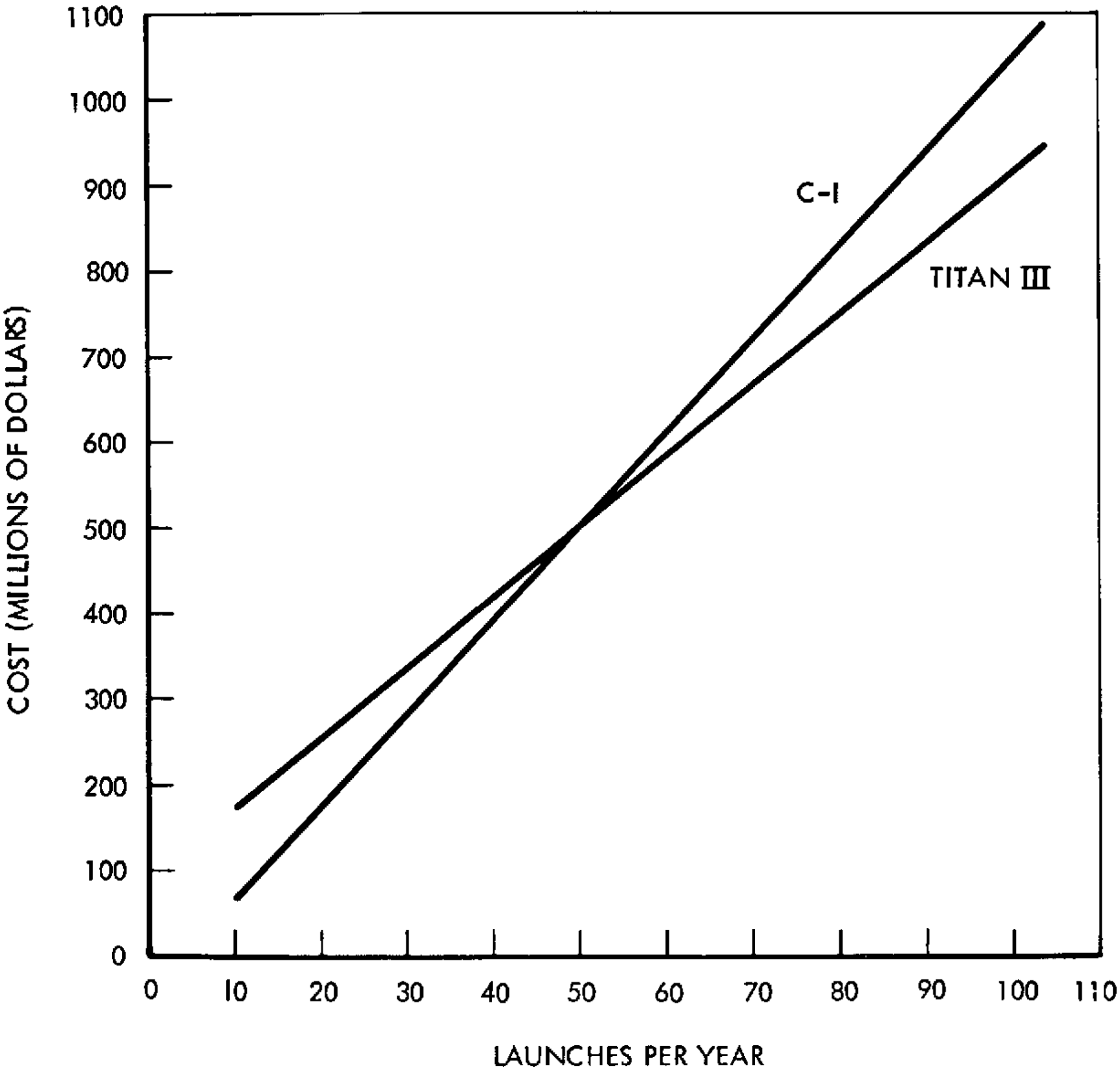


Figure 124. Total System Cost Factor for C-1 Booster Versus Titan III Booster

Employing the manned vehicles indicated in each case, initial launch of three space stations results in these initial manning factors:

1. Apollo (three-man) = 3 x 3 = 9 crewmen

2. Apollo (five-man) = 5 x 3 = 15 crewmen

3. Multimanned Personnel Transport (12-man) = 12 x 3 = 36 crewmen

4. Multimanned Personnel Transport (21-man) = 21 x 3 = 63 crewmen

The application of these ground rules to the particular reentry vehicle/booster combination results in the cost figures which are presented in summary form in Table 40. The costs in Table 41 for the individual reentry vehicles and boosters were used as a basis for the data shown.

It can be seen from these analyses that the most efficient personnel transport/resupply system for the space station operations—over a reasonably long period of time—is a new 12-man personnel transport launched by the Saturn C-1 (or by the Titan III). The cost of using the 21-man personnel transport with the Saturn C-5 is just slightly more. Even though the Saturn C-5 is grossly oversized for the 21-man vehicle, it is cheaper to use than to develop a new 50,000-pound payload booster (e.g., the Saturn C-3) for this particular program. In either case, however, the use of the new multiman personnel transport is considerably more economical than the use of the Apollo (basic or modified for five men).

These costs, although preliminary in nature, do point out major trends in the economics of space station operations. Further analyses must be conducted in order to refine these costs to include other variables such as ground support equipment, launch facilities, etc.

Table 40. Five-Year Operations Cost Comparison

Case No.	Vehicle-Booster System Combination	Total Launches Required*	Manned Vehicle Costs** ($)	Booster System Costs ($)	Total Operations Costs ($)
I	Apollo/C-1 3-Man (not reusable)	715	2180×10^6	7110×10^6	9290×10^6
Ia	Apollo/C-1 3-Man (reusable)	715	1358×10^6	7110×10^6	8470×10^6
II	Apollo/C-1 5-Man (not reusable)	434	1270×10^6	4150×10^6	5420×10^6
IIa	Apollo/C-1 5-Man (reusable)	434	844×10^6	4150×10^6	4994×10^6
III	Multimanned Personnel Transport/C-5 21-Man (reusable)	104	514×10^6	2500×10^6	3014×10^6
IV	Multimanned Personnel Transport/C-3 21-Man (reusable)	104	514×10^6	3650×10^6	4164×10^6
V	Multimanned Personnel Transport/C-1 12-Man (reusable)	208	618×10^6	2068×10^6	2686×10^6

*5 percent Attrition accounted for
**Reusable vehicles figures include refurbishment cost

Table 41. System Cost (Millions of Dollars)

System	Number of Systems									
	5	9	10	13	18	20	26	35	52	104
Multimanned Personnel Transport (Reusable)										
Research and Development			298.4			298.4				
Production and Operation			116.9			233.8				
Total Cost			415.3			532.2				
Apollo Capsule (Not Reusable)										
Production and Operation	22.4	36.3			65.1		89.7	116.4	174.4	320.5
Total Cost	22.4	36.3			65.1		89.7	116.4	174.4	320.5
C-1 Booster										
Production and Operation	62.3	105.2			197.1		274.2	361.2	527.0	1034.1
Total Cost	62.3	105.2			197.1		274.2	361.2	527.0	1034.1
Titan III Booster										
Research and Development - Stage I	125.0	125.0			125.0		125.0	125.0	125.0	125.0
Production - Stage I	19.1	30.8			53.7		72.8	92.4	127.6	227.1
Production - Stages II and III	19.3	30.1			54.9		74.1	94.5	130.0	227.8
Operation	6.9	14.8			36.7		59.4	87.5	146.6	364.0
Total Cost	170.3	200.7			270.3		331.3	339.4	529.2	943.9
C-3 Booster										
Research and Development				1840.4			1840.4			
Production and Operation				246.6			450.4			
Total Cost				2083.0			2290.8			
C-5 Booster										
Production and Operation				311.0						
Total Cost				311.0						2500.0

TYPICAL EXPERIMENTS

Most experiments that are currently being conducted in space are seriously hampered by weight limitations. In a number of notable examples, the results of experiments conducted in these unfavorable circumstances have led to a great amount of confusion in regard to the true interpretation of the resulting data. With the advent of a large, manned space station, sufficient weight and volume become available to conduct a great number of highly sophisticated experiments in the "space laboratory". The availability of crew members to monitor the experiments and to set up new tests makes the station extremely versatile and greatly enhances its value.

Concurrent with the next phase of the space laboratory development, exhaustive studies of probable experiments should be made to determine equipment weights, volume power requirements, etc., and to establish optimum experiment schedules, manpower requirements, and minimum equipment requirements. While it was not possible to conduct extremely detailed analyses in the 6-month feasibility study, a number of experiments were tentatively defined in an attempt to obtain an approximation of required weight and power allocations. Table 42 describes a number of typical experimented and associated requirements.

Since it will not necessarily be required to conduct all experiments simultaneously—and thus provide for all related equipment and supplies at the time—an accurate assessment of weight and power allocations is not now possible. On the basis of analyses to date, however, an arbitrary allocation of 20,000 pounds and approximately 1.5 kilowatts of power (peak) for scientific experimentation appears reasonable.

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Table 42. Estimated Requirements for Typical Experiments

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Electronics					
Vehicle Recognition:	1550	50	3400	3	6 months
a. Radar with various receivers, transmitters, etc.	1000	35	3000		
Radar test equipment	300	8	100		
b. I. R. system	160	4	250		
I. R. test equipment	70	2	20		
c. IFF equipment	20	1	30		
Electrostatic Charge Equalization, Static Charge Detector, Equalization Probe	8	0.2	3	1	1 day
Communications Testing, RecordStorage Paper, etc.	10	0.2	0	1	2 weeks to 3 months
Prop. of Light Communication Link, Light Source and Optics	50	1.0	500 (avg)	1	2 weeks to 3 months
High Intensity Light Source Transmitter, Light Source and Optics	50*	1.0*	500*	1	2 weeks to 3 months
High Sensitivity Photo Detectors, Photo Detectors and Optics	50	2.0	50	1	2 weeks to 2 months
Communications Using Millimeter Waves, Transmitter and Omnidirectional Antenna	2.5	0.1	10	1	2 days

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Radiation Device Tests, Millimeter Wave Transmitter	10	0.2	20	2	2 days
Communications Using Infrared Waves, Infrared Source and Optics	50	1.0	500 (avg)	1	1 week to 1 month
Communications Using UHF Waves (1700 to 2300 mc), Transmitter and Omnidirectional Antenna	2.5	0.1	10	1	1 week to 1 month
Communications Using UHF Waves (225 to 400 mc), Transmitter and Omnidirectional Antenna	2.0	0.1	5	1	1 week to 1 month
Analysis of Frequency Spectrum to Determine the Extent of the Interference Problem:	40	1.3	100	2	1 week to 2 months
a. Variable Frequency Receivers	20	0.5	70		
b. Directional Antenna	20	0.8	30		
Communication System Reliability, Paper Storage, Recorder	5	0.1	2	1	2 weeks to 6 months
Application of RF Techniques for Space Communications	2	0.1	5	1	1 week to 1 month
Passive Communications Relays, Reflector Arrays (Folded State)	120	2	5	1	2 weeks to 2 months

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Particle Communication, Accelerator and Radiation Detection	500	10	10,000	2	1 week to 2 months
Van Allen Radiation Effects	10 210*	0.2 2.1*	0 5*	1	2 weeks to 2 months
Magnetic Effects on Communications System: a. Sensitive magnetometer b. Mass spectrometer and detector package	1595*	11.75*	410*	1	2 weeks to 2 months
Radiation Sensors to Determine Earth's Vertical: a. IR horizon scanner b. Microwave scanner	15 10 5	0.5 0.3 0.2	10 5 5	1	1 week
Vehicle Velocity Meter Accuracy Test, Stellar Abberascope	20	0.5	10	1	1 day
Target Surveillance, Detection, and Identification: a. Radar and IR devices	 1530*	 49*	 3370*	2	6 months

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Fabrication and Test of Large Physical Antennas:	353	2.05	5	1	1 month to 3 months
a. Unfurlable antenna	350	2	0		
b. Receiver/ transmitter	3	0.05	5		
Testing of Fix Taking Devices and Techniques:	408	10.2	1200	2	2 weeks to 2 months
a. Radar	90	2.2	200		
b. Terrain mapping radar	108	2.8	600		
c. Local vertical-stellar inertial	160	4.0	300		
d. Position computer	50	1.2	100		
General Reliability Testing:	10	0.2		1	1 month to years
a. Paper storage, etc.	10	0.2	0		
b. Guidance system	135*	3.1*	250*		
Guidance System Suitability Test:	135	3.1	250	2	1 month to years
a. Stellar-inertial-optical system	105	2.4	200		
b. Reference guidance system	30	0.7	50		
Tracker Test (Cold Body), Radar and IR Devices	1530*	49*	3370*	2	2 weeks to 6 months
Surveillance Radar Techniques, Radar and Test Equipment	1300*	43*	3100*	1	1 week to 1 month

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Electro-Optical Techniques, Camera, Optics, Electronics and Test Equipment	160	4.0	115	1	1 week to 1 month
Optical Scanners:	230	5.5	820	1	1 week to 1 month
a. Earth moon sextant	10	0.2	10		
b. Star trackers	10	0.2	10		
c. Wide angle trackers	10	0.2	10		
d. Horizon scanners	10	0.2	10		
e. Sun trackers	10	0.2	10		
f. Star abberascope	10	0.5	20		
g. Inertial platform with optics mounted	80	2.0	400		
h. Electronics, computer, power supply	80	2.0	350		
Optical System Evaluation:	170	4.1	300	1	2 weeks to 2 years
a. Reference system (optical)	170	4.1	300		
b. Optical scanners	230*	5.5*	820*		
Guidance System Evaluation, Reference and Test Guidance Systems	105	2.4	200	1	2 weeks to 2 years
Inertial Component Test:	100	2.6	185	1	1 week to 2 months
a. 3-gyro inertial platform	30	0.75	100		
b. Platform electronics	10	0.25	40		
c. Optical theodolite	10	0.3	5		
d. Computer, storage and T/M	25	0.6	40		
e. Maneuver fuel	25	0.7	0		

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Gyro Drift Tests, Platform, Computer, Storage, and Optical Theodolite	75*	2.3*	185*	1	1 week to 2 months
Long Baseline Radar System Experiment:	33	1.1	210	1	2 weeks to 6 months
a. Atomic clock	30	1.0	200		
b. Transceiver	3	0.1	10		
Radio Astronomy Measurements of Cosmic Noise:	510	0.3	12	2	2 months
a. Antenna (dipole or phased array)	500	External			
b. Receiver and data recording system	10	0.3	12		
Radio and Radar Astronomy Measurements of Solar Flares:	200	8.0	100	2	2 weeks to years
a. Radar system	200	8.0	100 (avg)		
b. Radio system	510*	-	12*		
Radio Astronomy Measurements of Cosmic Noise in SNF Range (15 to 50 KMC):	25	0.7	25	2	2 months
a. 2.5 ft diameter antenna	10	0.3	10		
b. Receiver and data recording system	15	0.4	15		
c. Radar (for earth atmosphere only)	175*	3.0*	170*		

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Structures - Materials					
Polymerization of Elastomic Monomers by Radiation, Small Can	5	1.0	0	1	1 month
Evaluation of Elastomers in a Combined Environment of High Vacuum and Radiation Derived from Ultra Violet, X Ray, Gamma and Primary Cosmic Rays:	5	1.0	0	1	1 month
a. Turnbuckle on stand	4	0.7	0		
b. Balance scale	1	0.3	0		
Fluid and Gas Permeability of Various Elastomers:	55	5.0		1	1 month
a. Materials	50	4			
b. Small Cans	5	1	0		
Effects of Weightlessness on Performance of Lubricants and Heat Transfer Fluids:	1000	42.0	100	2	1 day to 6 weeks
a. Pumps and fluids	100	5			
b. Devices to evaporate, boil, and condense	100	4			
c. Various thermodynamic cycles	500	15			
d. Lubricant and heat transfer fluids	50	3			
e. Structures and instrumentation	250	15			

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Bearing Lubrication at Zero Gravity:	70	6.0	200	2	6 months
a. Bearing test rig	20	3.0			
b. Lubricants and heat transfer fluids	50	3.0			
Degradation Measurements for Plastics, Light Transmission Sensing Elements	5	1	100	1	1 month
Test of Structural Materials for Long Time Operation Under High Vacuum Condition	5	1	0	1	3 months
Environmental Correlation:	135	9		2	1 week to 2 months
a. Cans to collect decomposition	20	1	0		
b. Equipment to determine outgassing and meteoroid damage	15	3			
c. Equipment for heat transfer characteristics measurements	100	5			
Radiation Heat Transfer:	150	10	100	1	1 month to 2 months
a. Heat power and temperature-time measurements	50	5	100		
b. Equipment for heat transfer characteristics measurements	100	5			

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Repair and Application of Coatings in Space:	100	12	200	2	1 week to 6 months
a. Spray tank, compressor motor	50	10	200		
b. Materials and coatings	50	2	0		
Meteoroid Damage(s) of Coatings:	10	1		2	2 weeks to 6 months
a. Microscopic reflectometer	10	1			
b. Equipment to determine outgassing and meteoroid damage	15*	3*			
c. Equipment for heat transfer characteristics measurement	100*	5*			
Structural Resistance to Meteoroid Impact and/or Penetration	0	0	0	2	6 months
Effect of the Structure Upon Radiation Particle Energies, Cosmic Rays, etc.:				1	1 week
a. Detector package	10*	0.6*	6*		
b. Signal processing	78*	1.5*	60*		
Measurement of Cosmic Ray Effects and Interaction with Matter:				1	2 months
a. Detector package	10*	0.6*	6*		
b. Signal processing	78*	1.5*	60*		

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Photovoltaic Cell Evaluation of Various Cells	10	1.0	0	1	6 months
Photovoltaic and Photoemissive Material Deterioration Evaluation	10	1.0	0	1	6 months
Pressure Loss Through Structural Organic Material:	45	11.0		2	2 weeks to 3 months
a. Pressure gages	5	1.0			
b. Pressure systems for airlock	40	10.0			
Decompressional Fatigue of Rubber Mechanical Items:	5	1.0	0	1	3 months
a. Small devices					
Exposure to Environment:	205	6.0	100	2	6 months
a. Meteoroid Detector	5	1	100		
b. Various heat protection materials	200	5	0		
Test Methods and/or Techniques of Joining Materials and Components, Rivet Gun, Welding Equipment and Hand Tools	50	10.0	200	2	6 months
Space Repair Techniques and Occupational Problems	10	3.0	0	2	6 months

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Determination of Fabrication and Sealing Techniques and Heat Transfer Patterns for Structural Parts, Hand Tools, etc.	10	3.0	200	2	6 months
Radiation Thermal Equilibrium Studies:	85	16.0		2	2 weeks to 6 months
a. Skin temperature, emissivity and absorption measurement	10	3	100		
b. Equipment to determine energy balance	50	10			
c. Equipment to determine thermal environment	25	3			
Thermal Environment:				2	1 week to 1 month
a. Equipment to determine energy balance	50*	10*	100*		
b. Equipment to determine thermal environment	25*	3*			
c. MTSS equipment	0	0			
Thermal Strains:	20	3		2	1 week to 1 month
a. SR4 strain and temperature recording	20	3			
b. Equipment to determine energy balance	50*	10*	100*		

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Thermal Strains: (Cont.)					
c. Equipment to determine thermal environment	25*	3*			
Fatigue Crack Propagation in a Vacuum	5	1	0	1	1 month
Space Vehicle Thermal Equilibrium Study:	1220	52		3	1 week to 1 year
a. MTSS equipment	0	0			
b. Various Radiator designs	150	10			
c. Equipment to determine energy balance	50	10			
d. Advanced environmental control system	1000	30			
e. Temperature control system	20	2			
Performance and Reliability Data of Secondary Environmental Control System:	1150	39	1000	3	6 months to 1 year
a. Liquid to gas converters	80	4			
b. Mixed fluids in single container	50	3			
c. Advanced environmental control system	1000	30			
d. Temperature control systems	20	2			

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Performance of Surface Coatings in Space Environments:	20	4		2	6 months
a. MTSS equipment and coatings	5	1	0		
b. Equipment to determine outgassing and meteoroid damage	15	3			
Energy Balance Effect of Vehicle Geometry:	110	17	100	1	6 months
a. MTSS equipment	35	4			
b. Equipment to determine energy balance	50	10			
c. Equipment to determine thermal environment	25	3			
Cooling Methods:	1150	40	1500	1	6 months
a. MTSS equipment	0	0			
b. Various radiator designs	150	10			
c. Advanced environmental control system	1000	30			
Noise and Vibration Test:	50	10	20	1	6 months
a. Instruments					
<u>Biomedical</u>					
Weightless Condition Feeding Devices:	116	9		2	1 hour
a. Devices	10	1			
b. Food and packages	106	8			

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Adjustable High Accelerations, Long Term Comfort Crew Seat Effectiveness Study	15	0.5		2	1-28 days
Tests of Personnel Restraint Equipment	21.5	0.5		1	3 days
Feasibility Test of Special Bed for Use Under Weightless Conditions	5	3.0		1	3 days
Evaluation of Instrument Displays	300	40	10	3	1 day
Emergency Crew Protection and Escape Sub-system	1000	External		2	3 hours
Evaluation of Ground Training Techniques for Space Crews				2	2 days
Suitability Test of Training Simulators and Devices	50	60	10	2	3 hours
Feasibility of Using Space Station for Training, Training Simulators and Devices	90*	60*	10*	3	5 hours
Validation of Hypnosis as a Technique for the Development and Maintenance of Human Performance Capability, Training Simulators and Devices	90*	60*	10*	2	1 week

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Man-Machine Contribution to Test Operation Reliability				1	1 week to 1 month
Psychophysiology of Space Flight	309	10		3	6 months to 1 year
Habitability Test:	30			3	6 months to 1 year
a. Recording equipment	30				
b. Training devices	90*	60*			
Body Dynamics Under Zero Gravity:	10			2	7 days
a. Monitoring equipment	10				
b. Training devices	90*	60*			
Evaluation of Function of Space Workers Suit Assembly	96	27		2	3 hours
Study of Nonpressurized Environmental Clothing	40	4		1	10 days
Muscle Tone in Space	6	2		2	2 weeks
Physical Fitness of Man in Space	6*	2*		1	2 weeks
Metabolic Study of Man in Space	5	5		1	10 days

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Study of the Combined Stress of Thrust Acceleration Transition Zone to Weightlessness and Following Weightlessness to Circulatory, Respiratory, and Central Nervous System	200	70	10	2-3	1 month
Human Reactions to Weightlessness	200*	70*	10*	2-3	5 months
Study of the Stress of Reentry and Impact Deceleration on the Circulatory, Respiratory, and Central Nervous System of a Subject Who Was Previously Exposed to Weightlessness of Long Duration	1000*	70*	10*	2-3	5 months
Human Performance During Long Term Space Flight Conditions	200*	70*	10*	2-3	5 months
Insect Metamorphosis in Space	1	1		1	6 months
Baker's Yeast Growth and Fermentation in Space	1	1		1	1 day
Micro-Organism Exposure to Space Environment	2	1		1	2 weeks
Hydroponic Vegetation Reproduction Tests	10	1		1	2 weeks

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
<u>Propulsion - Power</u>					
Propellant Storability Test, Temperature, Pressure, and Vibration Instruments	50	2	10	2	1 month to 1 year
Zero Gravity Rocket Thrust Changer, Propellant Ignition	150	6	100	3	3 days
Propellant Expulsion Bladder Tests, Temperature, Pressure and Vibration Instruments	50*	2*	10*	1	4 days
Propellant Tank Slosh and Vortexing	150	10	10	1	2 days
Solar Thermionic Static Power Generating System (500 watt size)	50	1		1	3 months
Communication Power Sources:	240	6		1	1 week to 3 months
a. Solar source	50*	1*			
b. Chemical source - batteries	40	1			
c. Nuclear source	200	5			
Solar Mechanical Closed Loop Rankine Power Generation System (3 kw size)	811	2	30	2	3 months
a. Sunflower I system	746	External			
b. Meters and controls	65	2			

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Environment Multi Engine Start and Restart, Spent Final Booster Stages	50	2		4	1 week
Test of an Electric Propulsion Unit, Propulsion Unit and Instrumentation	50	1	1200	1	2 days
Ion Propulsion and Recombination Propulsion Unit and Instrumentation	50	1	1200	1	2 days
Plasma-Electromagnetic Interactions, Plasma Jet Unit	50	1	1200	1	2 days
Power Supply System Development Tests, Various Power Supplies	415	10.4	2	2	2 weeks
Test of Propellant Actuated Device Components in Space Environment:	100	15.5	200	1	1 week to 1 year
a. Pressure-force instrumentation	10	0.3	5		
b. Current-time instrumentation	5	0.2	5		
c. Strand burner	85	15	190		
Reaction Control Devices:	50	2	10	2	10 days
a. Temperature, pressure and vibration instrumentation	25	1	5		
b. High speed movie camera	25	1	5		

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Nuclear Reactors:	2110	23		3	2 months
a. Meters	100	20			
b. Load devices	10	3			
c. Power system	2000	External			
Space Environment					
Analysis of Perturbations of a Space Station, 3 Accelerometers (Orthogonally Mounted)	10	0.2	20	1	1 month
Ionization in Vicinity of a Space Station, Double Sphere Plasma Probe	30	0.5	1	1	2 weeks
Properties of the Ionosphere:	186	4.3	229	1	3 months
a. Mass spectrometer	30	0.6	40		
b. Detector package	10	0.6	6		
c. Magnetometer	3	0.1	3		
d. Radio sounder (receiver)	65	1.5	120		
e. Signal processing	78	1.5	60		
Measurement of Velocity Distribution Function of the Electrons Present in Outer Space, Electrostatic Energy Analyzer	20	0.8	10	1	2 months
Experiments Concerning the Dynamics of Particles, Ionization Changer and Associated Equipment	30	2	15	1	3 months

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Radio Astronomy Measurements of Van Allen Radiation Belts:	210	2.1	5	1	6 months
a. Sensitive receiver	10	0.1	5		
b. Unfurlable antenna	200	2.0	0		
The Detection and Dosimetry of Radiation in Space:				1	6 months
a. Detector package	10*	0.6*	6*		
b. Signal processing	78*	1.5*	60*		
Indirect Probing of Electromagnetic Properties of Planetary Atmosphere and Interplanetary Space Conditions:	1405	57.5	180	1	1 year
a. Telescope	1200	50	0		
b. Attachments	30	4.5	10		
c. Radar (active and passive)	175	3.0	170		
Direct Probing, Sensing, and Sampling of Planetary Atmospheres and Interplanetary Space Conditions via Spacecraft Installed Sensor Equipment, Detectors, Spectrometers, Ionization Chambers	236*	7.1*	254*	1	3 months

*Indicates items duplicated in other equipment

Table 42. Estimated Requirements for Typical Experiments (Cont.)

Title and Equipment	Weight (lb)	Volume (ft^3)	Power (watts)	Manpower for Test	Duration of Test
Direct Probing, Sensing, and Sampling of Physical Properties of Planetary Atmospheres and Interplanetary Space Conditions via Expendable Devices, Plasma Probe and Electrostatic Analyzer	50*	1.3*	11*	1	3 months
Observatory for Celestial Observations, Telescope and Attachments	1230*	55*	10*	1	6 months

*Indicates items duplicated in other equipment

V. REFERENCES

STRUCTURES AND MATERIALS

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